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ADVISORY GROUP FOR AERONAUTICAL RESEARCH AND DEVELOPMENT

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SOME LOW-SPEED PROBLEMS OF HIGH-SPEED AIRCRAFT

by

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SUMMARY

The first part of the paper deals with the low-speed aerodynamics of aircraft shapes suitable for achieving a required range at supersonic speeds. No attention is given to 'slewed' wings, nor to possible application of powered lift or variable geometry.

Wind tunnel tests are described on a simplified model with boundary layer control methods applied. Mention is also made of the possibility of adverse ground effect on maximum lift.

The second part of the paper is concerned with work aimed at clarifying some of the requirements for handling qualities of future aircraft.

Flight tests on an Avro 707A aircraft, with artificially worsened characteristics, are described, and it is shown that substantially constant performance in the piloting task can be achieved at the expense of increased pilot effort. Some tentative conclusions on desirable levels of speed stability and phugoid damping are drawn.

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NOTATION

c	wing chord
$\bar{\bar{c}}$	aerodynamic mean chord $\left(= \int_{-b/2}^{+b/2} c^2\, dy \div \int_{-b/2}^{+b/2} c\, dy\right)$
b	wing span
s	wing semispan
l	overall length (equal to centre-line chord for a slender wing with unswept trailing edge)
t	maximum thickness of a streamwise wing section
S	wing area; for a swept wing, defined by producing leading and trailing edges to centre-line; for a slender wing, the gross plan area.
p	planform parameter, S/bl
A	aspect ratio, b^2/S or $2s/lp$
ϕ	angle of sweepback
α	angle of incidence
β	angle of sideslip
δ	flap angle, deflection measured along wind
ξ	aileron angle
q	rate of rotation in pitch
C_D	drag coefficient (= drag/$\frac{1}{2}\rho V^2 S$)
C_L	lift coefficient (= lift/$\frac{1}{2}\rho V^2 S$)
ΔC_L	change in lift coefficient caused by flaps
C_m	pitching moment coefficient (= pitching moment/$\frac{1}{2}\rho V^2 S\bar{\bar{c}}$)
C_l	rolling moment coefficient (= rolling moment/$\frac{1}{2}\rho V^2 Sb$)
C_n	yawing moment coefficient (= yawing moment/$\frac{1}{2}\rho V^2 Sb$)
m_q	dynamic derivative (= $\frac{1}{2}\partial C_m/\partial(q\bar{\bar{c}}/V)$)

l_v	static derivative $(= \partial C_l/\partial\beta)$
n_v	static derivative $(= \partial C_n/\partial\beta)$
Δl_v	change in l_v caused by incidence
Δn_v	change in n_v caused by incidence
l_ξ	aileron power, $\partial C_l/\partial\xi$
C_μ	momentum coefficient, defined using total wing area
V	free-stream velocity
f	frequency
n	frequency parameter $(= f\bar{\bar{c}}/V)$
F(n)	spectrum function
a, b	constants in expression for lift of a slender wing
A, B	constants in expression for pitching moments of a slender wing
μ_1	relative density parameter $(= W/\rho S\bar{\bar{c}})$
W	weight of aircraft
ρ	air density
Δx	distance of centre of non-linear lift ahead of centre of linear lift

Stability axes are used throughout.

SOME LOW-SPEED PROBLEMS OF HIGH-SPEED AIRCRAFT

A. Spence* and D. Lean**

PART 1

Low-Speed Aerodynamics

1.1 INTRODUCTION

The high-speed aircraft whose low-speed aerodynamic problems are discussed in this part of the paper belong to the future rather than to the past or present. Küchemann[1] has shown how jet propulsion and the use of a new set of aerodynamics appropriate to supersonic speed lead one from the classical aircraft to new shapes suitable for achieving a required flight range. These shapes include wing-body arrangements with wing sweepback angles of 55° or 60° suitable for a Mach number of about 1.2, and slender, near-triangular wings with sharp leading edges suitable for Mach numbers of around 2 or more, depending on the ratio of span to length.

The low-speed investigations described here on swept-wing and body arrangements started by examining the possibility of using a type of flow with separation from the leading edge. It is shown that low-speed characteristics force one to aim at fully attached flow, and this is the main basis for the choice made in Reference 1 of this type of flow. Simplified pilot experiments are described in which use was made of blowing from the knee of downward-deflected leading-edge flaps together with blown trailing-edge flaps. Mention is made of the possibility of adverse ground effect on maximum lift.

More space is devoted to the case of slender wings suitable for a Mach number of 2 or more because clear advantages are derived from the new type of flow with separation from all edges. The steadiness of this flow is discussed from the point of view of the possibility of buffeting. Pitching-moment data are analysed, showing the effect of planform on low-speed aerodynamic centre and pitch-up. Model test results on static lateral and directional stability are used to show the effect of the wing geometry on rolling moments due to sideslip, and to yield a certain measure of understanding of the side force contribution to the yawing moments. Finally the effect of proximity to the ground is discussed and shown to be generally favourable.

No attention is given to 'slewed' wings, nor to possible applications of powered lift or variable geometry.

The last paragraph of the introduction of Küchemann's paper[1] is quoted; it applies here equally and could scarcely be better phrased - "it is gratefully acknowledged that many of the findings reported here are the results of a concerted and highly

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*_United Kingdom_

**_United Kingdom_

successful research effort on a very large scale, in which most British aircraft firms and research establishments have been engaged over the last 3 or 4 years. The responsibility for the presentation and conclusions is, of course, the author's".

1.2 AIRCRAFT WITH 55° - 60° SWEPT WINGS

1.2.1 Investigations on Models Having Separated Flow

Figure 1 illustrates a few upper surface flow patterns which have been observed at moderate incidence on wings of RAE.101 section, three of them having straight trailing edges, constant chord over the inner half of the span and parabolic variation of the chord between mid semispan and the tip as suggested by Bagley[2]. The fourth wing was designed to have constant local lift coefficient across the span according to a calculation by Brebner. The tests were made by Brebner at the R.A.E.[3] and by Garner at the National Physical Laboratory[4].

Figure 1a shows flow separation near mid-chord on a 9% thick wing of 60° sweep giving rise to a vortex lying above the rear part of the wing. Figure 1d shows leading edge separation on a 7½% thick wing of 46° sweep, but the flow is complicated by the spanwise flow over the rear part of the inner wing meeting streamwise flow behind the leading edge vortex further out. Figure 1b and c show simple leading edge separation continuous over the whole span at very moderate incidences on wings of thickness and sweep most relevant to our present purpose.

Pitching moments for three of these wings and for a flat plate model with a fuselage, tested by Kirby at the R.A.E., are collected in Figure 2. These show the usual phenomenon of initial nose-down pitch on swept wings of aerofoil section, when the separation is confined to the area near the tip and gives non-linear lift there; this is followed by a change to pitch-up as the separation extends inboard and there is a large loss of lift near the tip. The flat plate model showed only pitch-up from a low incidence without the preliminary pitch-down.

This led to a further investigation by Kirby in which flat plate wings of varying aspect ratio and taper were combined with a body and a tailplane which could be fitted at various heights above and below the plane of the wing. A few of the results are shown in Figure 3. With praticable tailplane heights, pitch-up is avoided only by having an aspect ratio as low as 2; for aspect ratio 2.7, there is marked pitch-up except for the case with the tail 0.[illegible] wing spans below the wing. In addition, with a sweptback central fin fitted, all the models became directionally unstable above incidences of 15° or 20°.

The other effect of flow separation on 55° swept wings which has been investigated is that on the performance of plain trailing edge flaps. Results are shown from tests by Trebble at the R.A.E.[5] on an untapered 55° swept wing with a sharp leading edge fitted with full span trailing-edge flaps. Although the flap lift increment on unblown flaps at moderate angles was not very sensitive to incidence, when the flap angle was increased to 60°, and particularly with blowing applied from the flap shroud, the lift increment fell sharply above 5° incidence. In the light of later results on slender wings, it seems likely that vortex breakdown was occurring and causing this drop. Deflection of a leading edge flap and blowing from its shroud towards the leading edge was tried in an attempt to control the flow, but this did not produce the improvement desired, possibly because the blowing slot was too far from the edge or because the angle of sweep of the leading edge was too small for this technique to be applied successfully.

From the above and other investigations, it was concluded that it would be more promising to try to avoid separated flows on these wings.

1.2.2 Tests and Problems on Swept Wings Having Attached Flow

A series of tests was made in the de Havilland Low-Speed Tunnel[6] on a half model untapered wing of aspect ratio 3 and sweep 55°. For speed and economy the section had a sharp leading edge and was of constant thickness from 0.2c to 0.8c. 20% chord leading-edge flaps and trailing-edge flaps were fitted, with tangential blowing at the knee in both cases. Full-span trailing-edge flaps, especially when blown, gave intolerable pitching-moment characteristics as expected and will not be discussed further. A few results with inner half span trailing-edge flaps at 60° are shown in Figure 5. The best results obtained were with full span or inner half span leading-edge flaps deflected 30° or 45° down, and with blowing at the knee of these and of the trailing edge flaps at their respective critical C_{μ}'s . Increasing the leading-edge flap angle from 30° to 45° and the span to full span increased the stalling angle, but usually at the expense of a small reduction in lift at moderate incidence. Except where boundary-layer control was applied at the knee of the leading-edge flap there was early pitch-up, but this was delayed almost to the incidence for maximum lift by leading-edge boundary-layer control.

Taken together with other British and American experiments, the results show that boundary-layer control at the knee of leading and trailing edge flap works well even with hinge-line sweeps as high as 55°, and that the usable incidence can thereby be raised to at least 20°, whilst lift increments not much below potential flow values are obtained*. As for the blowing momentum requirements, it is suggested that the values of C_{μ} necessary in the de Havilland tests (about 0.015 and 0.025 for half span trailing and leading-edge flaps respectively) were unduly high because of the crudity of the wing section used. It is worth adding that the use of suction rather than blowing at the knee of both leading and trailing edge flaps could well halve the flow quantities required.

Turning to one or two other problems of low-speed aerodynamics on these designs, it is clear that with angles of sweepback of 55°, rolling moments due to sideslip will be high, particularly on the approach. Although this problem has received little attention, it appears possible that provision of boundary layer control over the outer half of the leading edge may have to be faced in order to maintain sufficient aileron power to cope with cross-wind landing and side gusts.

Finally, attention is called to the possibility of adverse ground effect on maximum lift. The effect of the ground is equivalent to that of an image of the aircraft reflected in the ground. The bound vorticity of this image causes a reduction in the effective longitudinal velocity at the aircraft proportional to the lift co-

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*In a subsequent experiment on the same model, the leading-edge flap was reduced to 0.1c and had a rounded edge whilst the trailing-edge flap chord was increased to 0.3c. This test gave disappointing results which are by no means understood.

efficient and varying inversely as the height above the ground. Unless this effect is offset by a favourable redistribution of lift, maximum lift is adversely affected and evidence in plenty, ranging from 'high lift' models of 20 years ago to the Comet and later designs, can be adduced for issuing the warning that the probability of a loss in maximum lift should be borne in mind.

1.3 SLENDER WING AIRCRAFT

Much of the early research in the field of slender wings has been done on relatively simple models fulfilling the requirement of fixed primary separation lines along continuous, sharp leading edges. Attention is mainly confined in this paper to this class of models which are either nominally flat plates or have simple cross-section shape, usually rhombic. The planforms range from the gothic with parabolic leading edge to the simple delta and to the ogee with streamwise tips and at least one point of inflexion in the leading edge. Throughout the work referred to here, the trailing edge is straight and unswept; this has at any rate one virtue, that of simplicity.

For convenience, use is made of two of the geometrical parameters employed by Küchemann[1] and many others:

$\frac{s}{l}$ = the semispan-length ratio

p the ratio between the wing area and the area of the circumscribed rectangle with the same span and length ($= s/2sl$)

1.3.1 Flow Steadiness, Buffeting

One of the most desirable qualities of the flow round an aircraft in flight is that of steadiness, so that the aircraft is free from buffet. Maskell and Weber (Reference 7 and elsewhere) and others have argued eloquently that slender wings with sharp edges and regular development of leading-edge vortices give this steady flow.

T.B. Owen and his group at the R.A.E.[8] and Dr. J.P. Jones[9] at Southampton University have investigated the aerodynamic excitation on delta wings. Jones's work was mainly directed to studying panel flutter in the high-speed case. Owen has studied surface pressure fluctuations on a flat plate 70° delta, and the low-frequency content of the normal force fluctuations on rigid models of delta wings of plano-convex cross-section over a range of sweepback[8]. The results of the latter experiments are summarized in Figure 6, where the incidence giving a certain magnitude of aerodynamic excitation, $\sqrt{(n\,F(n))}$, at frequency parameter $n = f\bar{\bar{c}}/V = 0.05$ is plotted against sweepback angle. For a given wing, this low-frequency fluctuation remains low until the incidence is increased to that at which vortex breakdown occurs in the immediate vicinity of the wing. The excitation then rises rapidly, indicating that severe buffet is probable. Figure 6 shows that the incidence involved are above about 19°, 27° and 35° for wings of 65°, 70° and 76° sweep respectively, and other work shows that at 70° sweep, 10° sideslip reduces the limiting incidence by about 8°. by causing earlier vortex breakdown on the windward side.

Slender wings seem therefore to be free from the possibility of buffet over the practical incidence range at low speeds provided that sweepback angles of about 70^{o} or more are employed. Checks are needed on planforms other than the simple delta.

1.3.2 Aerodynamic Centre, Pitch-Up

A likely value of the rotary derivative m_q for a slender wing at low speed is about -0.2, and if one couples this with a value of $\mu_1 = 7$, typical of a large slender wing design, it is clear that the manoeuvre margin will exceed the static margin by only $0.03\bar{\bar{c}}$. Until artificial increase of manoeuvre margin by automatic means is acceptable not merely to make the piloting task easier but also to make it possible, the centre of gravity of such an aircraft must be placed close to the low speed aerodynamic centre. The problem of achieving trim at supersonic speed and the possible associated drag penalty drive the designer to minimize the low-speed stability margins, and so knowledge of the aerodynamic centre at low speed is even more vital than usual.

Following an early suggestion by the Handley Page Company, the available data are shown in Figure 7, simply plotted against the fore-and-aft position of the centre of area. Two straight lines are drawn, for flat plates and thick models respectively, thickness moving the aerodynamic centre back. More detailed examination of the original data shows that increase of semi span/length ratio has the opposite effect.

A rough analysis of lift and pitching-moment data for flat plate models and uncambered models of moderate thickness has been attempted, making the following assumptions:-

(a) that the lift can be expressed by

$$C_L = a\alpha + b\alpha^{5/3}$$

and (b) that the centres of the linear and non-linear lift remain fixed, independent of incidence, which gives

$$C_m = Aa\alpha + Bb\alpha^{5/3}$$

with a , b , A and B constant for a particular wing. These constants have been determined by fitting the data at $\alpha = 7\frac{1}{2}^{o}$ and 15^{o}. Although the method shows promise in the sense that the resulting curves fit the test points in general in the incidence range 0^{o} - 20^{o} within 0.005 on lift coefficient and 0.001 on pitching-moment coefficient, the results are not sufficiently tidy for presentation. However, it was considered worthwhile to plot the resulting distance between the centre of non-linear lift and that of the linear lift. Figure 8 shows this as a function of the position of the centre of area, though the planform parameter p would be equally good as there is almost a one-to-one correspondence between p and centre of area for the wings tested. Aspect ratio and effective taper would be expected to be of some importance, but the scatter in this very crude first analysis is too great for their effects to be distinguished. Figure 8 makes it clear that as incidence increases, wings of gothic planform pitch nose down, ogees with p less than 0.5 pitch up, delta wings pitch up more than corresponding wings with streamwise tips, and wings with their centre of area near 64% of their length behind the apex generally have straight pitching-moment curves.

More refined analysis on these lines may well prove fruitful in the absence of theoretical methods of calculating the loading when strong vortices are present; in this and in many other respects, however, Maskell's similarity theory of initial vortex development[10] promises to become a powerful tool.

1.3.3 Rolling Moments Due to Sideslip

Many experimenters have measured static lateral and directional stability derivatives on sharp-edged slender wings, generally without a fin fitted. It is found that the curves of rolling-moment coefficient against sideslip angle are straight until combinations of incidence and sideslip giving vortex breakdown ahead of the wing trailing edge are reached. The derivative $l_v (= dC/d\beta)$ is found to be roughly proportional to the lift coefficient and a useful summary of the available data is obtained by examining the values at a lift coefficient of 0.5 (which corresponds to an incidence of 15^o for uncambered wings of semispan/length ratio 0.25).

These values are collected in Figure 9 as a function of the ratio of the total length or centre-line chord of the wing to its span. With the exception of the results for the two cambered models, and of the high point for a very thick delta wing tested by Peckham[11], the results lie satisfactorily close to the curve drawn. There does seem to be a slight general tendency for the results on thick models to be higher than those for flat plates, but recent provisional results on cambered models suggest that $-l_v$ may increase or decrease depending on the type of camber.

A reasonable estimate for plain flap ailerons of chord 5% of the wing centre line chord extending from mid-semispan to the tip shows that these would give a value of l_ξ of about -0.06; so that some $3\frac{1}{2}^o$ of aileron would be required to trim out the rolling moment per degree of sideslip, for a wing of length-span ratio 2, allowance having been made for a central fin reducing $-l_v$ by 0.02. A good deal of effort has therefore been devoted to attempts to reduce $-l_v$, mainly by Kirby and his colleagues at the R.A.E. The results show two promising methods:-

(a) $7\frac{1}{2}^o$ dihedral over the inner two-thirds of the semispan with 15^o anhedral outboard reduces $-l_v$ by 0.05 on wings of $s/l = 0.2$ to 0.25, without affecting tip clearnace. The effect is roughly linear with the droop of the tip below the plane of the inner part of the wing.

(b) The addition of large, triangular extensions from the leading edge of the wing gave large reductions in $-l_v$, particularly if the extensions were drooped relative to the plane of the wing.

1.3.4 Yawing Moments Due to Sideslip

The yawing moment derivative $n_v = dC_n/d\beta$ for sharp-edged slender wings without fins is usually small and negative at zero incidence and becomes increasingly positive as incidence is increased. A detailed examination of the incidence effect has been made for two flat plates[11] and five thick models (from References 11 and 12 and unpublished work by Keating at the R.A.E.) and the results are shown in Figure 10. For the flat plate models, it is clear that the effect of sideslip is to produce a force normal to the wing with its line of action to windward of the centre line, since

$$\Delta n_v = - \Delta l_v \tan \alpha$$

On thick models, there is in addition the effect of the side force, the balance of which is usually into wind because the windward vortex approaches closer to the wing and causes increasing suctions on the forward and sideways facing upper surface. In general, the assumption of a constant fore-and-aft position for the line of action of the change of side force due to incidence gives good agreement with the data, as shown by the broken lines in Figure 10. These positions are given in the following table:

Planform	s/l	*Cross-section*	*Change of sideforce with incidence acts at*
Gothic	0.250	8% rhombic	$0.44l$
Gothic	0.333	12% rhombic	-
Delta	0.250	12% rhombic	$0.56l$
Delta	0.250	12% const. t/c	$0.60l$
Ogee	0.208	5% thick approx.	$0.49l$

The side force derivative on the thicker gothic wing is so very small that its line of action is indeterminate.

This analysis gives a reasonably clear picture of the combined effects of incidence and sideslip but there is ample scope for research to seek a clear understanding of many of the features.

The contribution of central fins on the classes of wings under consideration has been shown in several experiments[13] to be sensibly constant up to at least 25° incidence and to be in agreement with calculation assuming full reflection in the wing.

1.3.5 Ground Effect

The main effect on a slender wing of proximity to the ground is to cause a large increase in lift curve slope. Kirby at the R.A.E. has collected results from tests on several models and these are reproduced in Figure 11. For engineering purposes the results for different wing planforms collapse on to a single curve when the fractional increase in lift at fixed incidence is plotted against the height of the mean quarter-chord point in spans above the ground. 50 - 60% increase in lift is typical for a touch-down position.

Ground effect causes a nose-down pitching moment coefficient of about 0.02 at a lift coefficient of 0.5.

Other results may be briefly summarized:-

(a) the lift-drag ratio is constant at constant incidence

(b) rolling moments due to sideslip are constant at constant lift

(c) yawing moments due to sideslip are constant at constant incidence.

It may be concluded that ground effect on approach and landing is very favourable mainly because of the large increase in lift at fixed incidence. On take-off, this factor and the increase in lift/drag ratio at constant lift are very favourable, but this is offset to some extent by the nose-down pitching moment requiring increased up-elevator which decreases the lift.

1.4 CONCLUDING REMARKS

This part of the paper has attempted to show the present state of knowledge of the low-speed aerodynamic characteristics of 55° swept wings and of sharp-edged slender wings, based entirely on recent wind tunnel experiments.

On swept wings it is concluded that one should aim at attached flow, and pilot experiments on boundary layer control both at leading edge and trailing edge have been described. Further work of this kind is needed.

On slender wings, flow steadiness, longitudinal distribution of loading, pitch-up characteristics and static stability derivatives in sideslip have been discussed. More work is needed to reach a fuller understanding of almost all these features, but the low-speed problems look a great deal less formidable than on many older types of wing.

PART 2

Requirements for Handling Qualities

2.1 INTRODUCTION

The first part of the paper improves our understanding of the type of flow to be expected over some typical high-speed planforms, at incidences appropriate to low speed and landing, and illustrates the trends of some of the more important aerodynamic derivatives.

This section deals with work being done to clarify some of the requirements for handling qualities of future aircraft. As such, it is concerned not so much with forecasts of dynamic behaviour, etc., as with determining what behaviour the pilot wants, and what he can just tolerate.

There is, of course, a considerable amount of well-founded data on desirable handling qualities already in existence, which is continually being supplemented. It is clear, however, that there are certain gaps in the overall picture of the ideal aircraft which these handling qualities are supposed to define. This is particularly so in the case of low-speed and landing-approach behaviour, even for present-day aircraft. For example, current civil requirements[14] refer to the need for 'a satisfactory degree of speed stability, control of gradient of descent, and lateral and directional control ...... during the steady approach'. There are obvious difficulties in designing for compliance with this sort of requirement, and rectification of deficiencies after the prototype has flown can be very expensive and time-consuming.

We shall therefore deal mainly, in this part of the paper, with work being done to help clarify these requirements.

2.2 DESIRABLE AND ACCEPTABLE HANDLING QUALITIES

The development of automatic flight control and automatic landing systems may seem to make the need for precision in defining handling qualities less important. However, not all aircraft and airfields will be equipped for automatic landing, at least for some time, nor will the equipment be 100% reliable. We must, therefore, know the handling qualities of the basic aircraft in some detail, so that we can determine what design changes or automatic devices are required to bring these qualities up to the 'desirable' level. Equally, it is important to know whether, if automatic devices ('black boxes') *are* used, the handling qualities of the basic aircraft are above or below some minimum acceptable emergency level. If they are below that level, then the automatic system must be protected - e.g. by multiple redundancy - so that, in effect, it never fails.

There is thus, perhaps, as much interest in defining minimum acceptable handling qualities as in defining 'desirable' levels. The work described below has this need in view.

2.3 AIRSPEED CONTROL AND SPEED STABILITY - CONVENTIONAL AIRCRAFT

As is well known, flight at speeds below the minimum drag speed poses certain piloting problems if the aircraft is constrained to follow a particular straight path. Any change in airspeed from that at which the drag equals the thrust results in a speed divergence which the pilot must correct.

Reference 15 describes a study of the relation between the chosen approach speeds and the drag characteristics of 19 different conventional aircraft. It was concluded that speed stability is a factor to be reckoned with in choosing a minimum comfortable airspeed, and that the tolerable stability level depended on the type of approach being made. On a carrier approach, with a heavy premium on excessive speed, instability with a time constant of 20 seconds is accepted, for the relatively short approach commonly used. On a pilot-controlled instrument approach, lasting perhaps 5 times as long, positive stability with a time constant of 60 seconds or less is demanded.

The speeds used in this analysis were those commonly accepted as comfortable minima. The pilots were not required to investigate lower airspeeds, where speed stability would be worse, and, in any case, other aspects of handling, which may also deteriorate at lower speeds, have to be considered as well. In fact, in 50% of the cases, speed was limited by effects other than those related to speed stability.

The above procedure does not, in general, allow us to determine the point at which speed instability becomes intolerable, because of these other effects. The pilot's acceptance (or otherwise) of a certain level of stability must be coloured by the overall difficulty of controlling the aircraft, which determines the amount of attention he can devote to the speed control problem.

However, present indication are that if speed stability is no worse than that mentioned above, there should be no cause for serious complaint from the pilots. This is not to say that a higher degree of stability would not be appreciated, or that, in emergency, a worse level could be tolerated. To investigate this further, an aircraft with variable speed stability is required, so that tests may be made at a constant speed, and the effects of other speed-dependent variables eliminated.

2.4 TESTS WITH VARIABLE-SPEED-STABILITY AIRCRAFT

A prototype automatic throttle control system had been developed at the R.A.E., Bedford, as a means of improving speed stability on the approach, using an Avro 707A aircraft. Basically*, this equipment, fully described in Reference 16, used an airspeed transducer to modulate the engine thrust as speed deviated from the chosen datum. Then, if dT/dV is the change in thrust per unit change in airspeed, the equivalent change in drag coefficient is given by

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*In the tests described in Reference 16, various combinations of inputs were tried, including the integral of the airspeed error, pitch attitude error and incidence error, in order to obtain the best all-round performance under all conditions. For present purposes, we are mainly concerned with the airspeed error input.

$$(\Delta C_D)_E = -\frac{1}{\rho VS}\frac{dT}{dV} \qquad (1)$$

and the time constant, τ, of the subsidence of an airspeed error is

$$\tau = \left(\frac{V}{2g\dfrac{C_D + (\Delta C_D)_E}{C_L} - \dfrac{dC_D}{dC_L}}\right) \qquad (2)$$

Clearly, if $(\Delta C_D)_E$ is made sufficiently large, the aircraft can be changed from one with speed instability (negative τ) into one with positive speed stability of any desired level.

On the Avro 707A, many actual landings were made with airspeed gearings (dT/dV) in the range 120-180 lb/knot, on an engine with a maximum thrust of about 3500 lb. These produced time constants between +7 and +4 seconds in conditions where the basic aircraft had an (unstable) time constant of -10 seconds, with a welcome improvement in speed holding.

This same aircraft became available for a further study of the effects of speed stability. By simply reversing the sense of the airspeed transducer, the equipment was made de-stabilising. An additional de-stabilising input could also be used, which varied the engine thrust in response to changes in incidence, and simulated the effect of a high induced drag. A block diagram of the system is shown in Figure 12. It will be noted that the pilot's own demands for thrust changes are fed in electrically via an auxiliary throttle.

At the chosen datum speed of 120 knots E.A.S., the basic aircraft is reasonably pleasant to fly, is clear of the region of dutch-roll instability and is on the stable side of the minimum drag speed (τ = +80 seconds). At the maximum (de-stabilising) setting of the equipment, the (unstable) time constant is -4 seconds.

Many simulated instrument approaches have been flown, using an optical glide path defined by a theodolite alongside the runway. An operator passed angular glide path error information verbally to the pilot, via a radio link. Azimuth information was obtained visually. Each approach lasted 1½ to 2 minutes. Continuous records were made to glide path error, airspeed, elevator and throttle movements.

2.4.1 Qualitative Results

Recalling our previous suggestion that pilots on instrument approaches required positive speed stability, the results of these tests have been somewhat surprising. The pilots reported little or no deterioration in handling, in calm conditions, with an (unstable) time constant of -20 seconds, which was the previously reported comfortable limit for carrier landings. Even in 'very turbulent' conditions, landings have been made (*not* without complaint from the pilots) with time constants under -9 seconds - one pilot even coped with a time constant of -4 seconds on occasions, though this was exceptional. The pilots could not say what would be the minimum acceptable emergency level of speed stability, on the basis of these tests, but we must be cautious in assuming that what has been achieved here would be tolerable operationally, for the following reasons. These pilots were highly skilled, and in practice. The effects of learning were always present, though the test programme was arranged so as to eliminate this as far as possible. Simulation of instrument conditions was restricted to glide path information. Azimuth guidance by visual reference was easier and inevitably provided additional, unwanted, glide path information which some pilots admitted finding useful. Some approaches using Precision Approach Radar (P.A.R.) control proved to be significantly more difficult, particularly with time constants worse than -15 seconds, in that lateral corrections were often demanded by the ground controller at a time when the pilot was already busy with the speed control problem. When flying visually, the less-urgent lateral corrections could be made as convenient, as is also the case with carrier approaches using the deck landing mirror sight.

2.4.2 Analysis of Recorded Data - Airspeed Holding

Only a small sample of the recorded data can be presented here, though it is sufficient to illustrate the difficulties inherent in this class of experiment. Figure 13 shows the root mean square deviation of airspeed recorded on the worst of 6 approaches by each pilot at each of the discrete settings of the destabilizing equipment. These were chosen as being the nearest we could get to the results likely to be achieved by a pilot less practised in this exercise. Nevertheless, we must expect our results to be somewhat optimistic.

The significant feature of Figure 13 is the way in which airspeed errors show no great increase even in the most extreme conditions. A standard deviation of about 4 knots seems to cover all cases. Elevator movements seem equally undisturbed by increase in instability but the auxiliary throttle movements do become more violent in the worst cases. It must be noted that the pilots could see the main throttle lever moving (under the action of the servo) and could also hear the change in engine noise. These cues were in advance of, and drew attention to, larger airspeed changes. Large, rapid movements of the auxiliary throttle could, to some extent, be the result of the pilots trying to cancel the de-stabilising inputs of the experimental equipment.

The cross-correlation between elevator and throttle movements is very strong indeed (significance better than 1 in 1000). Thus whether pilots claim they use the elevator for airspeed control and throttle for glide path control, or vice-versa, the evidence is that they, in fact, use both controls together.

Analysis of these results is far from complete at this time, but one thing is clear - pilots' skill is effective in obscuring the effects of poor handling qualities. The need for an effective method of measuring how hard the pilot is working - mentally and/or physically - is equally evident.

2.5 SPEED STABILITY OF TYPICAL HIGH-SPEED AIRCRAFT

Without being specific about the configuration of such an aircraft, it will be sufficient, for purposes of illustration, to assume a typical lift/drag relation for the approach configuration, such as

$$C_D = 0.03 + 0.5\, C_L^2 \tag{3}$$

At a wing loading of 30 lb/ft^2, the minimum drag speed is 190 knots ($C_L = 0.245$). Figure 14 shows the variation, with lift coefficient, of the time constant of the speed stability. If an approach lift coefficient of 0.5 is to be used, the speed instability will have a time constant of 18 seconds. This would be quite unacceptable for normal instrument approaches, though it may be just tolerable in an emergency. Reduction of the lift coefficient to 0.4 (150 knots) would improve the emergency situation (τ = -30 seconds) but, clearly, an automatic throttle system will be required for normal operation.

If τ_1 seconds is the time constant of the basic aircraft, and τ_2 is the desired value, the airspeed 'gearing' (dT/dV) required is given by

$$\frac{dT}{dV} = -\frac{W}{g}\left(\frac{1}{\tau_2} - \frac{1}{\tau_1}\right) \tag{4}$$

Figure 15 shows the gearings required on this aircraft, for stable time constants of 60 seconds and 10 seconds, representing the minimum comfortable and probably desirable levels of speed stability. At a lift coefficient of 0.5, the higher level involves varying the thrust at a rate of 0.8% of the weight per knot, which is, proportionately, about half the rate used successfully on the Avro 707A in Reference 16. However, the absolute gearing - 1200 lb per knot on a 150,000 lb aircraft - is not insignificant, and in an actual design simulator studies would be required, using realistic engine response times, as well as studying the possible benefits of some of the additional control terms referred to in Reference 16.

A further situation in which speed stability may be a problem arises during the 'stacking' or 'loitering' phase of a transport aircraft. If we remove a profile drag increment of 0.025 due to the undercarriage, etc, the lift/drag relation becomes

$$C_D = 0.005 + 0.5\, C_L^2 \tag{5}$$

which gives a minimum drag speed (still at 30 lb/ft^2 wing loading) of about 300 knots. The 'loitering' speed may be nearer 200 knots, to fit existing traffic patterns, but, even so, the instability is not severe, having a time constant of about 70 seconds. However, since this speed may have to be held for much longer than the duration of an instrument approach, different rules may apply. In any case, control of height presumably need not be as tight as that required on an approach, in which case speed stability becomes less of a problem, but the damping of the phugoid oscillation - and associated height excursions - will need consideration.

2.6 GLIDE PATH CONTROL

This section is concerned only with the phugoid mode and its effect on glide path variability. Relatively much more is known about the effects of the short-period mode and elevator response. There are, however, some grounds for believing that the slow mode may be at least as important when height error inputs arrive at discrete intervals, as in a ground-controlled approach. Each discrete height correction, after the short-period response, can excite the phugoid oscillation. Even if the high error information is available more-or-less continuously, the pilot, with other problems to attend to, can only make discrete corrections, and if the phugoid is poorly damped, one would expect the height record to show some evidence of the presence of the oscillation. Reference 17 finds some evidence of phugoid-type oscillations in landing records of several large aircraft on ground-controlled approaches. Some of the glide path records in the present tests show traces of an oscillation which could be interpreted as the phugoid.

On the Avro 707A, the artificial changes in speed stability were, of course, accompanied by corresponding changes in phugoid damping*. Unfortunately, is was not possible to vary the effects independently. The vector diagrams for the flight path tangential forces show that the phasing of the two artificial vectors is such that their effects are roughly additive. This accounts for the nearly-unique relation that resulted between speed stability and phugoid damping, as shown in Figure 16, irrespective of whether the thrust was modulated by an airspeed input alone, or by airspeed and incidence changes together.

2.6.1 Flight Test Results - Glide Path Holding

The angle of elevation of the aircraft, as seen from the ground at the end of the desired glide path, was recorded and analysed at half-second intervals. The root mean square variation of this angle on the worst of 6 approaches made by each pilot at each discrete setting of the equipment, is shown in Figure 17. The base of this plot is the reciprocal of the time for the phugoid to double amplitude. It bears a very similar appearance to that of Figure 13, and shows that, as before, pilots' skill has masked the effect sought. They can maintain their performance of the task under deteriorating conditions at the expense of greater physical and mental effort.

On Figure 16 are marked the discrete conditions in which pilots first noticed a deterioration in handling, compared with the basic aircraft, as the instability was made more severe. In average turbulence, the first noticeable change occurred when the phugoid became neutrally stable, though we would hesitate to say that this was more than coincidence, at this stage.

2.7 SUMMARY OF TENTATIVE PROPOSALS FOR SPEED AND PHUGOID STABILITY

On the basis of these tests on the Avro 707A, which were concerned mainly with minimum acceptable handling, we feel that there is no case for reducing the desirable levels for normal operations. Speed stability should be positive, with a time constant less than 60 seconds. If an automatic throttle system is used, then a time constant of about 10 seconds seems a reasonable aim. Less is known about phugoid damping, but artificial improvement in speed stability will generally ensure adequate damping of the phugoid also. A time to decay to half amplitude of 40 seconds will probably be satisfactory.

---

*The relative damping ratio, E, is given, to a first approximation, by

$$E = \frac{C_D + (\Delta C_D)_E}{\sqrt{2}C_L}$$

where $(\Delta C_D)_E$ is the equivalent change in drag coefficient given in Equation (1).

In the emergency case (stability system inoperative) speed instability with a time constant of not less than 20 seconds may be just acceptable, provided that no other aspect of handling is only marginally acceptable at the same time. This speed instability could probably be just tolerated with a phugoid which doubled amplitude in, say, 50 seconds.

Experience has shown how pilots' skill can result in substantially constant performance in a worsening situation at the expense of increased effort. In establishing limits, reliance must therefore still be placed on pilot opinion, in the absence of an accepted means of measuring the pilots' work load.

Finally, in all work on the minimum acceptable level of a particular handling quality, it is important to include, wherever possible, the effect on that minimum of some variation of the other relevant handling qualities. This should be extended to areas which normally are investigated separately, e.g. longitudinal and lateral/directional. There may well be a significant 'stiffening' of individual requirements when the pilot has to deal with a situation in which all aspects of handling deteriorate together.

ACKNOWLEDGEMENTS

The authors gratefully acknowledge the assistance of Mr. D.A. Kirby in the preparation of the first part of the paper and of Mr. K.J. Staples in the preparation of the second part. The views expressed are those of the authors and do not necessarily represent those of the Ministry of Aviation.

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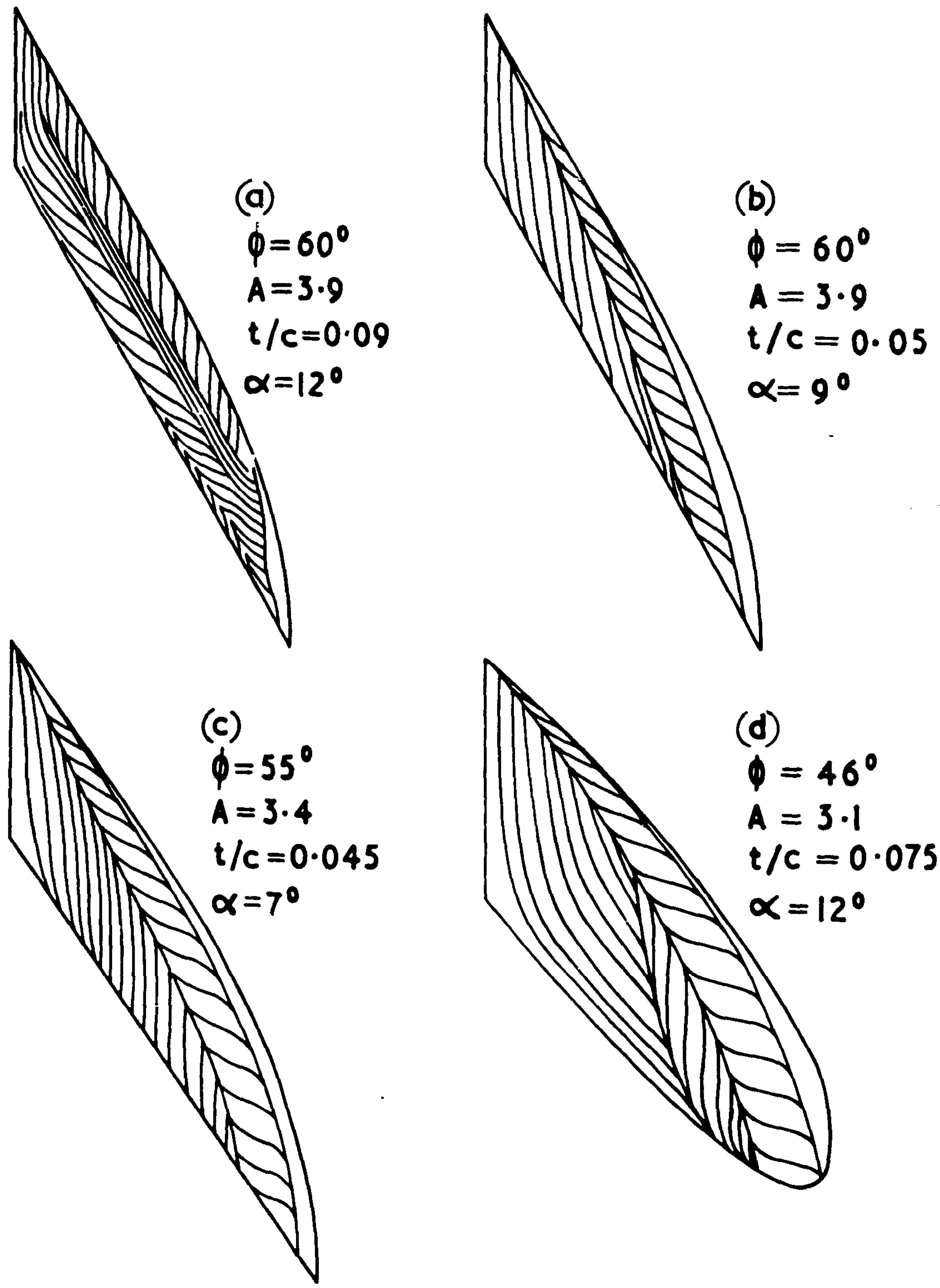


Fig. 1 Types of flow at high incidence on uncambered swept wings of R.A.E. 101 section

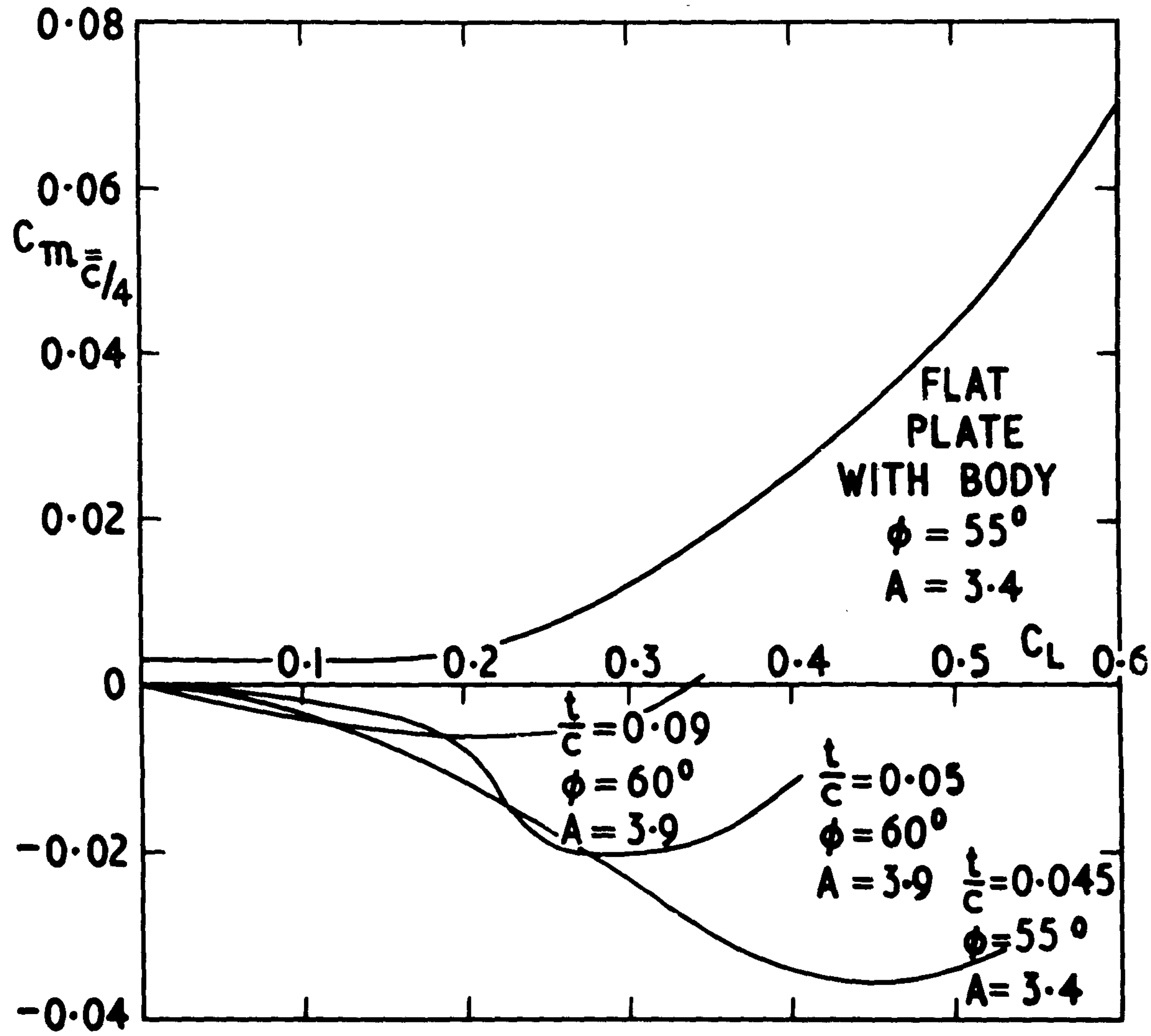


Fig.2 Pitching moments of swept wings

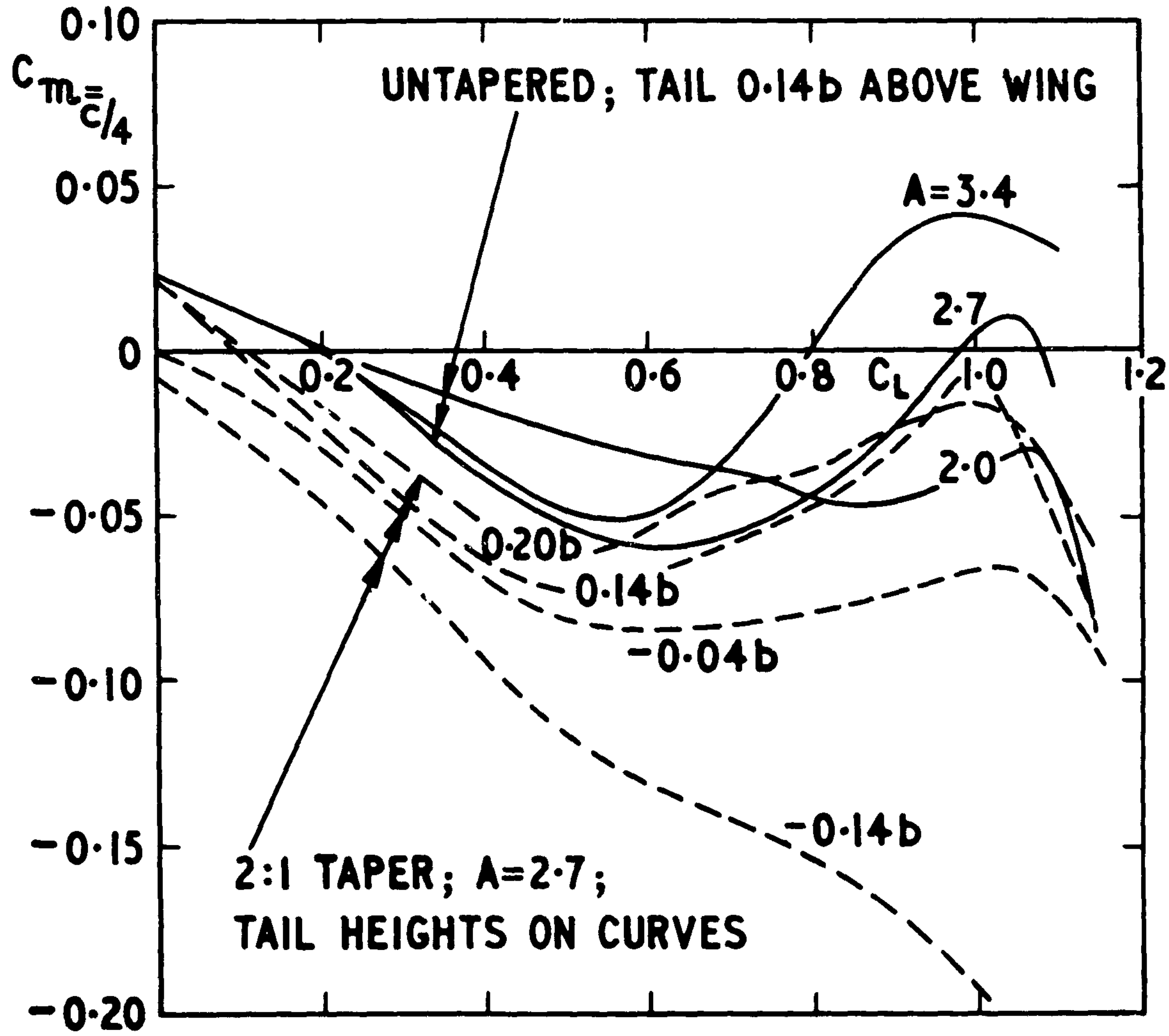


Fig.3 Effects of aspect ratio, taper ratio and tailplane height on pitching moments flat plate 55° swept wing and body

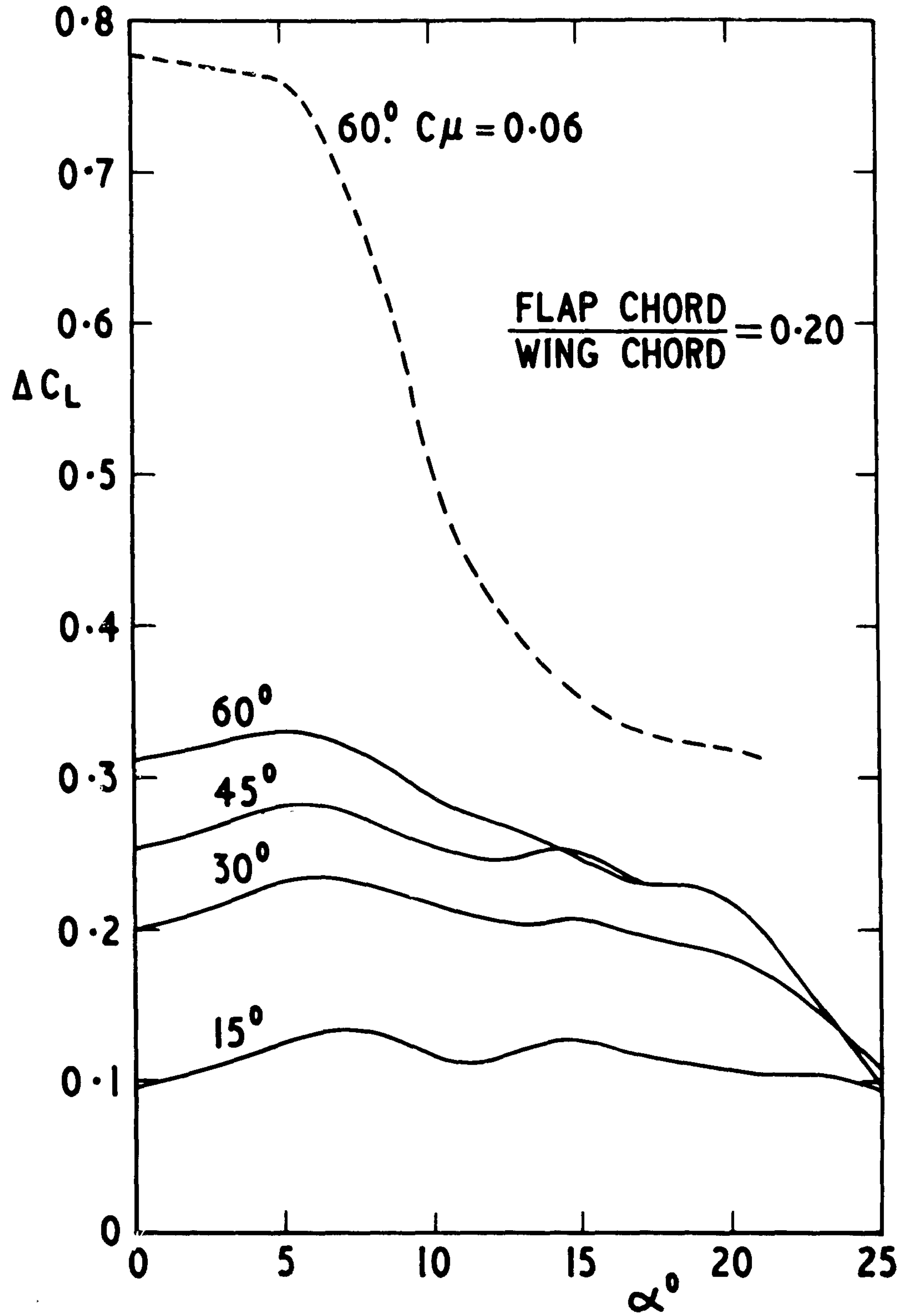


Fig.4 Effect of full-span T.E. flap on untapered 55° swept wing with sharp leading edge (A = 3)

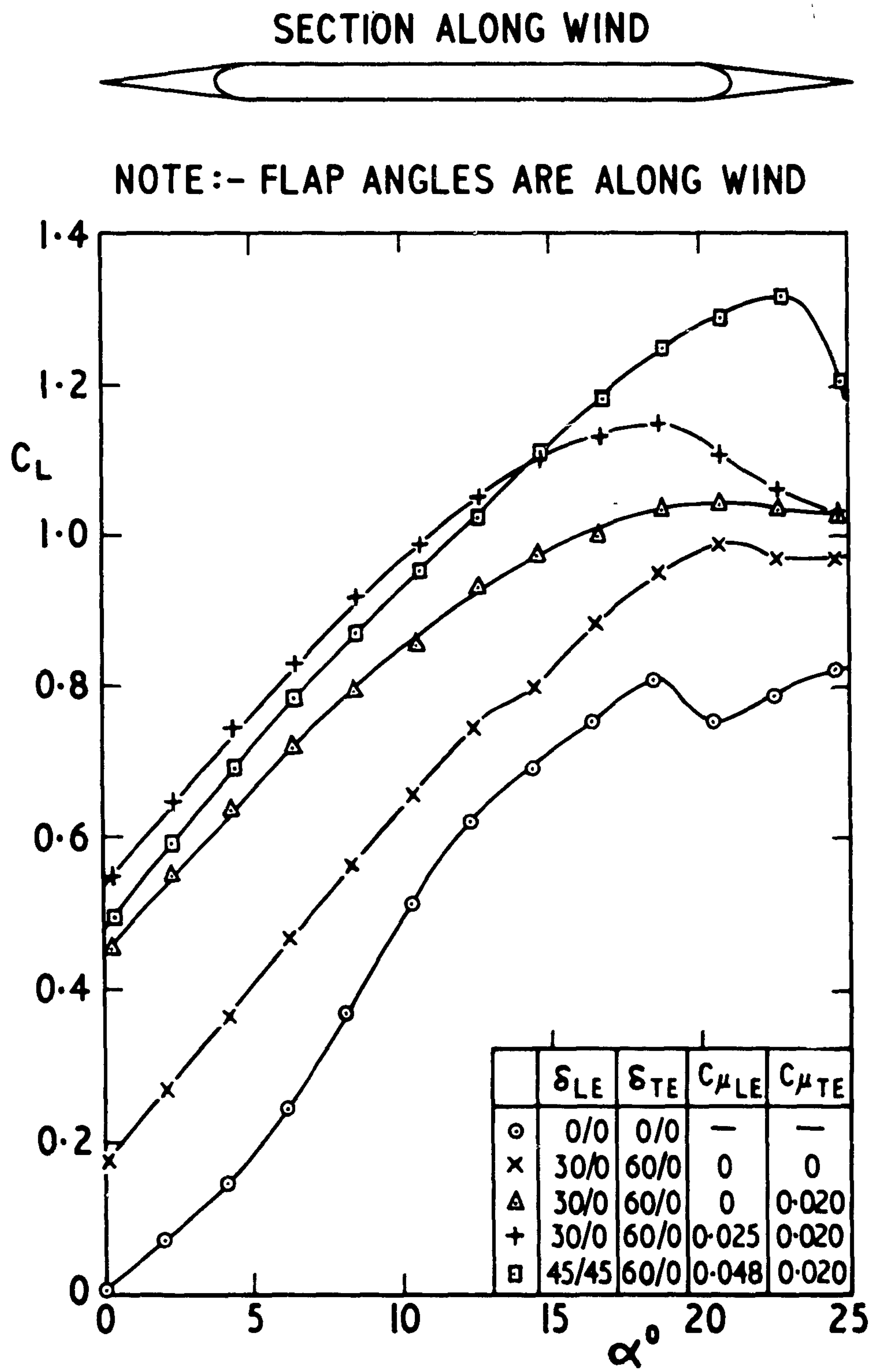
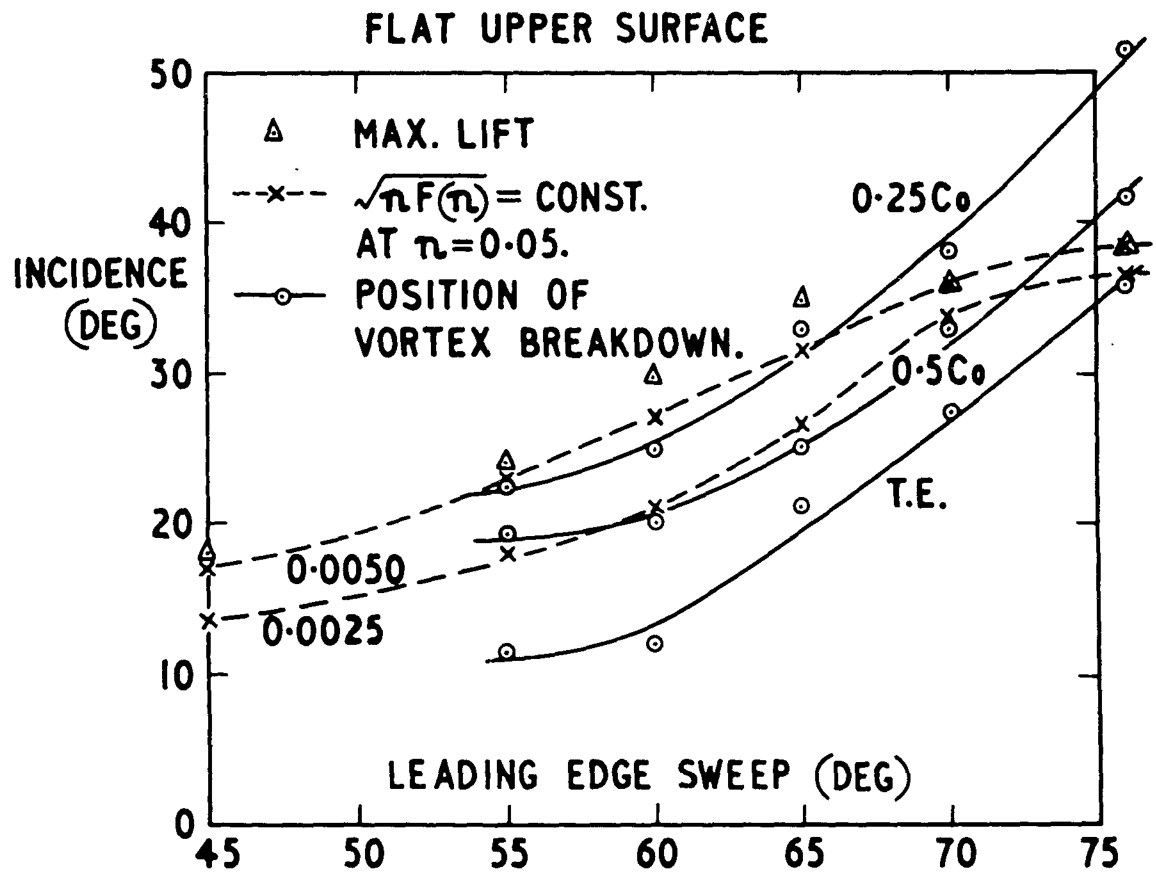
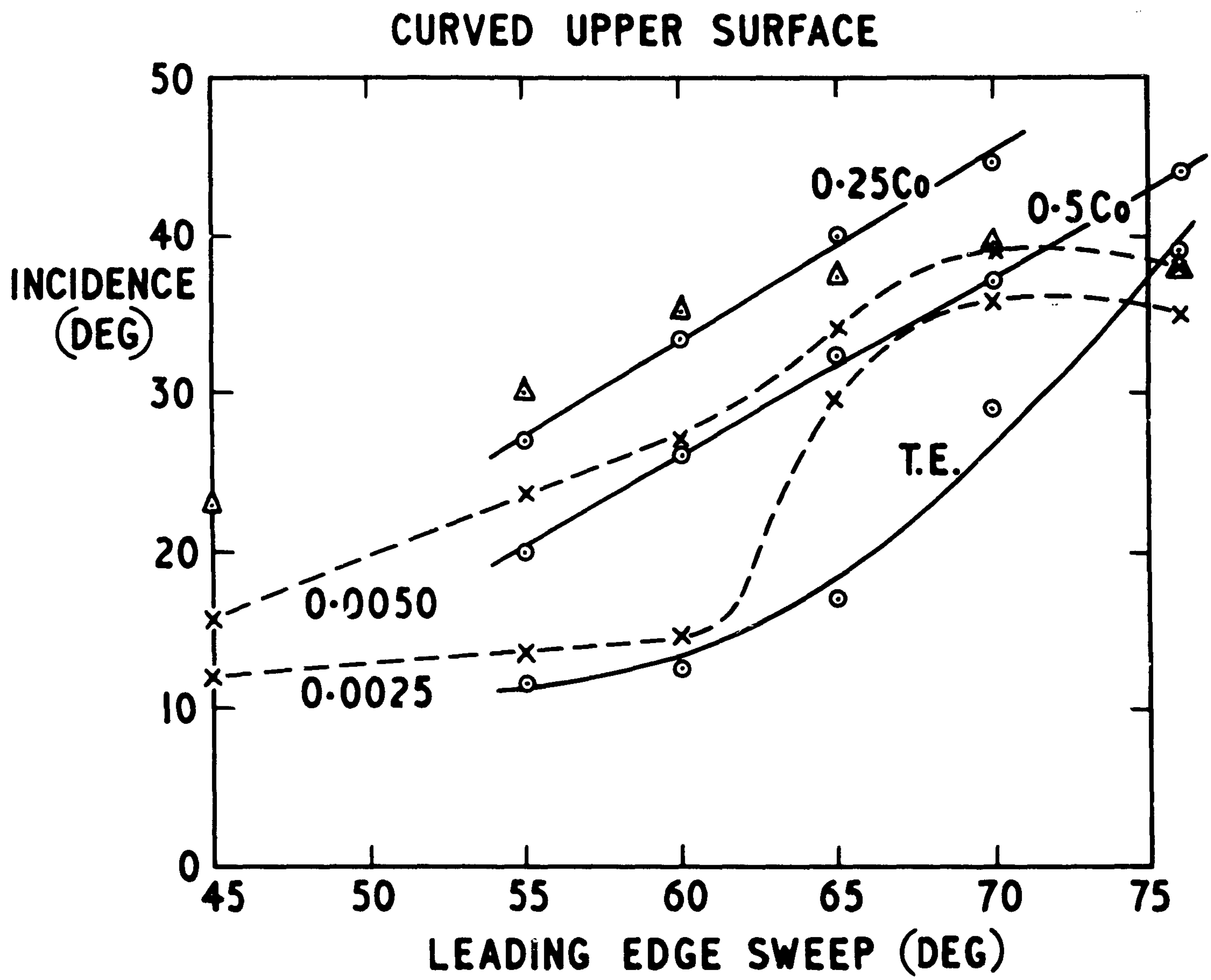


Fig.5 Blowing at knee of leading-edge and trailing-edge flaps; untapered, sharp-edged 55^0 swept wing



(a) Flat upper surface

Fig.6 Relation between position of vortex breakdown and normal force fluctuation on delta wings



(b) Curved upper surface

Fig.6 Relation between position of vortex breakdown and normal force fluctuation on delta wings

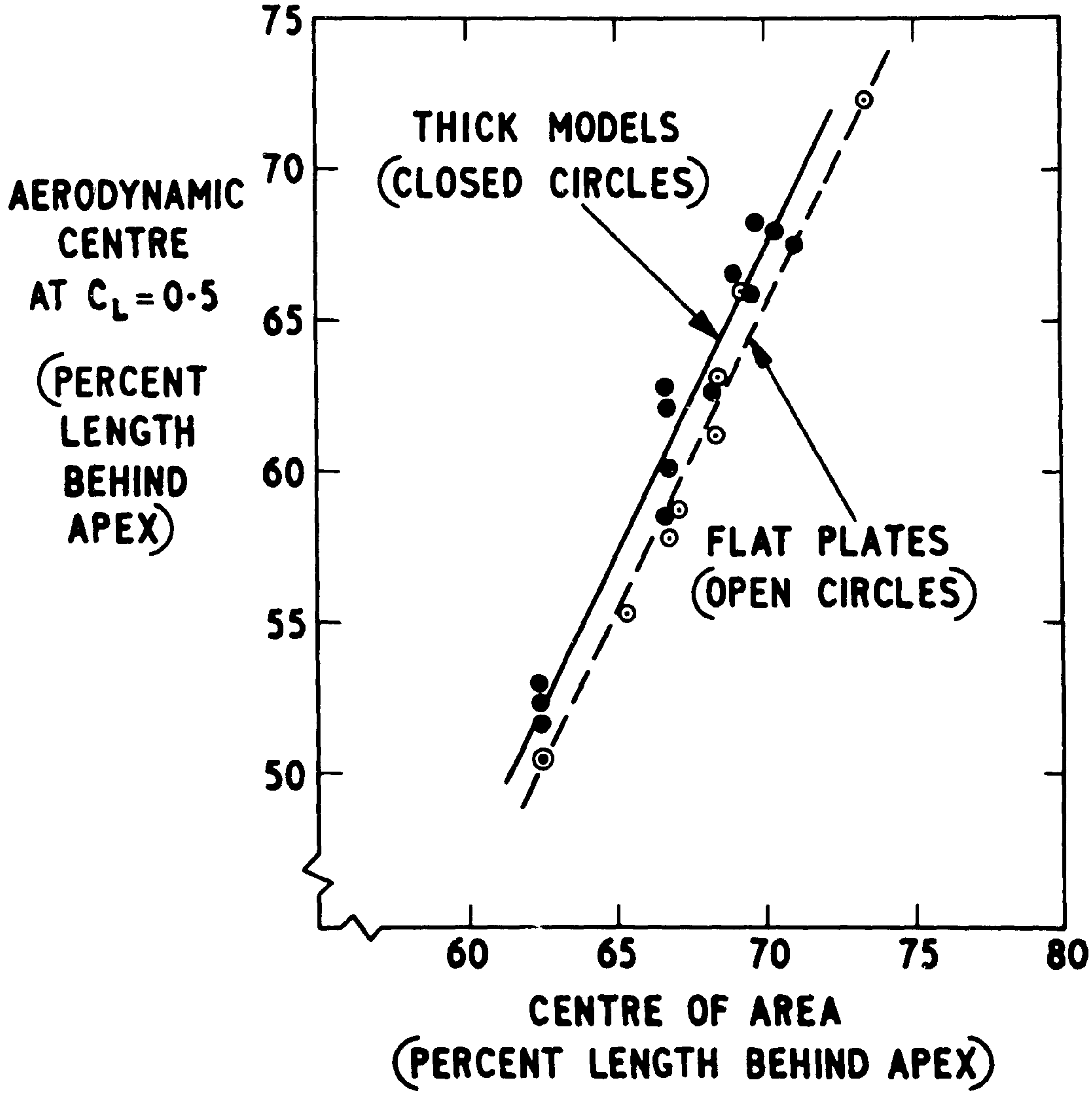


Fig.7 Aerodynamic centre of slender wings at $C_L = 0.5$

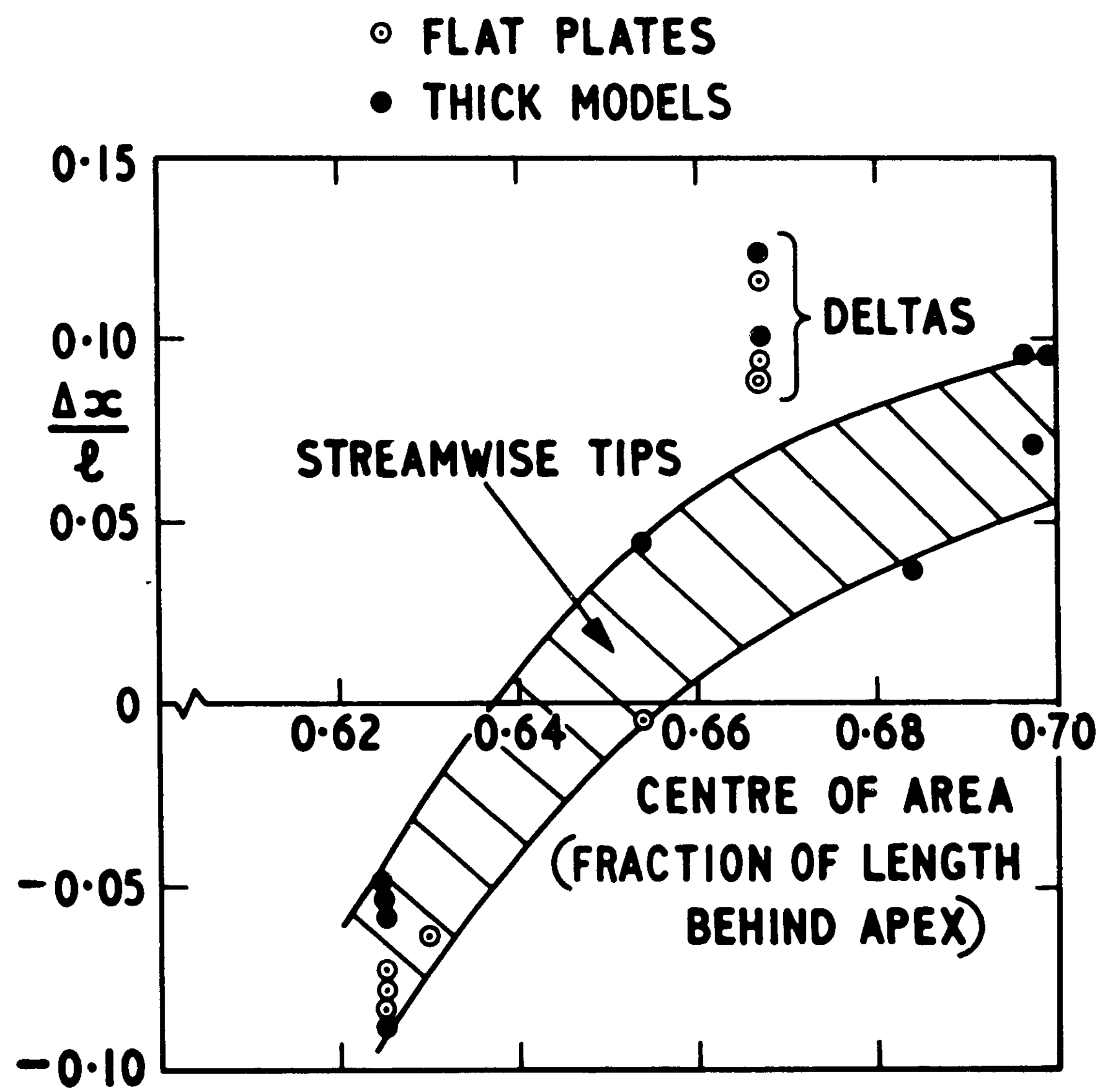


Fig.8 Distance of centre of non-linear lift ahead of centre of linear lift for slender wings

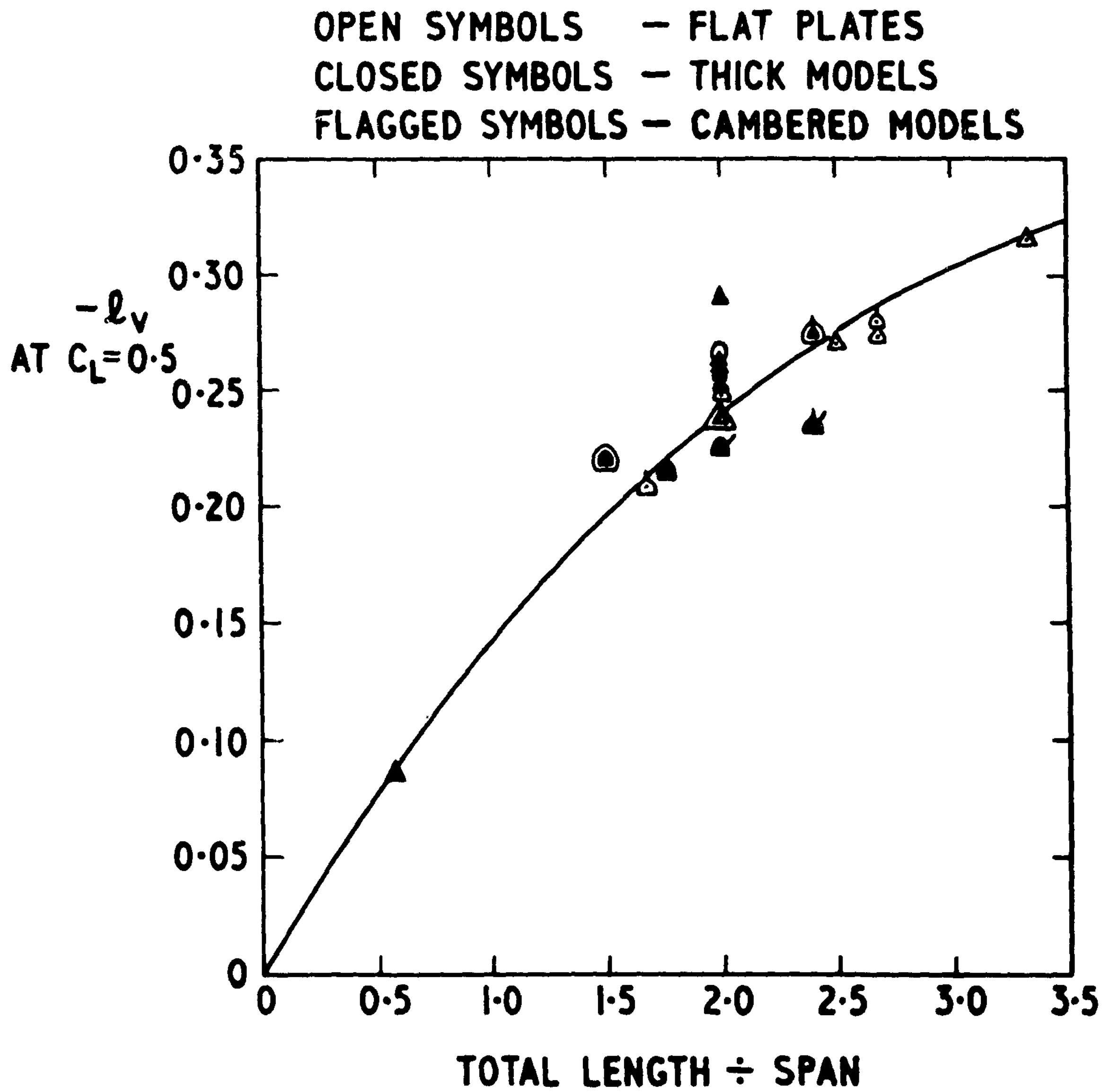


Fig.9 Rolling moment due to sideslip on sharp-edged slender wings at $C_L = 0.5$

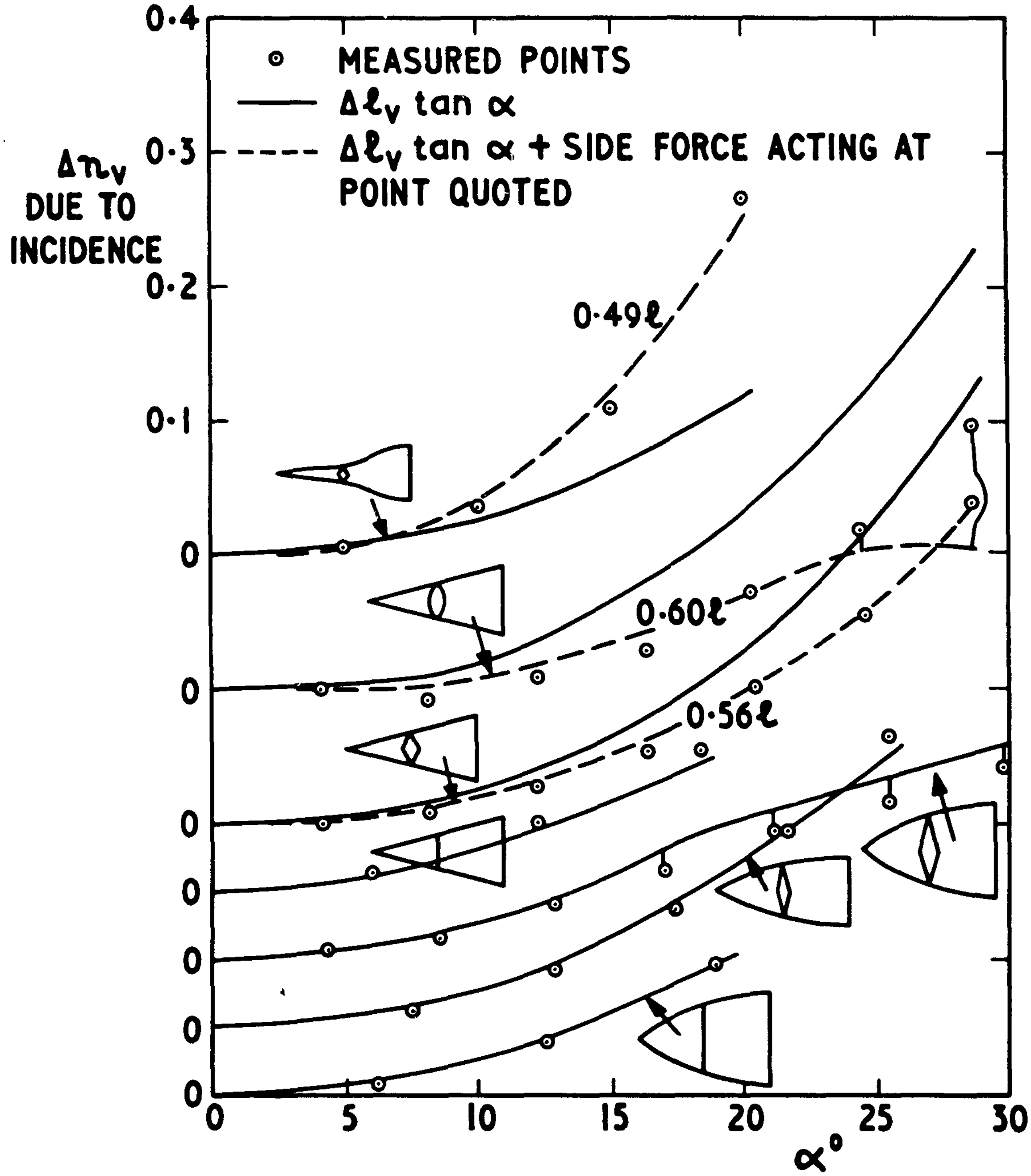


Fig.10 Incidence effect on static directional stability of slender wings

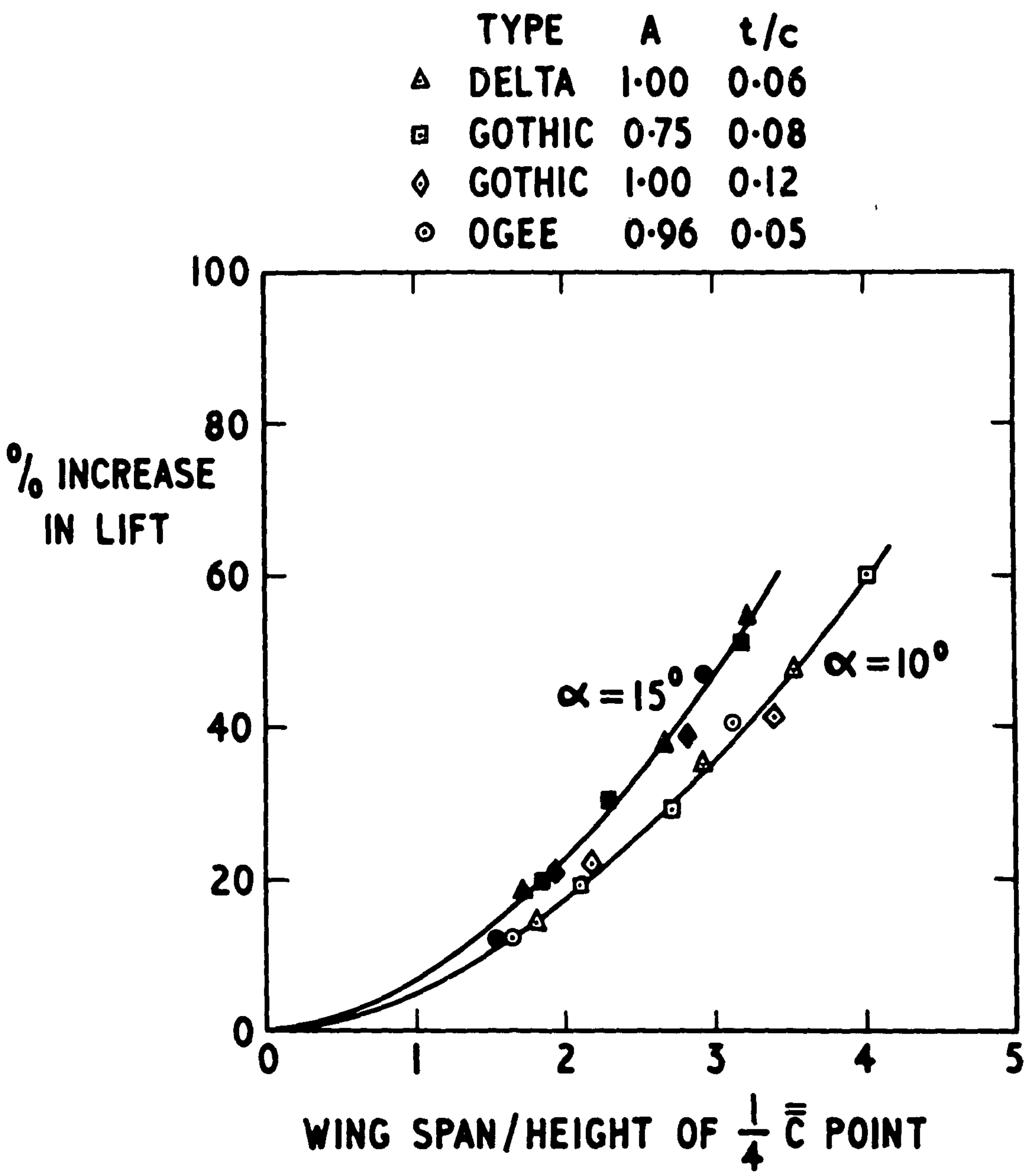


Fig.11 Ground effect on lift of slender wings

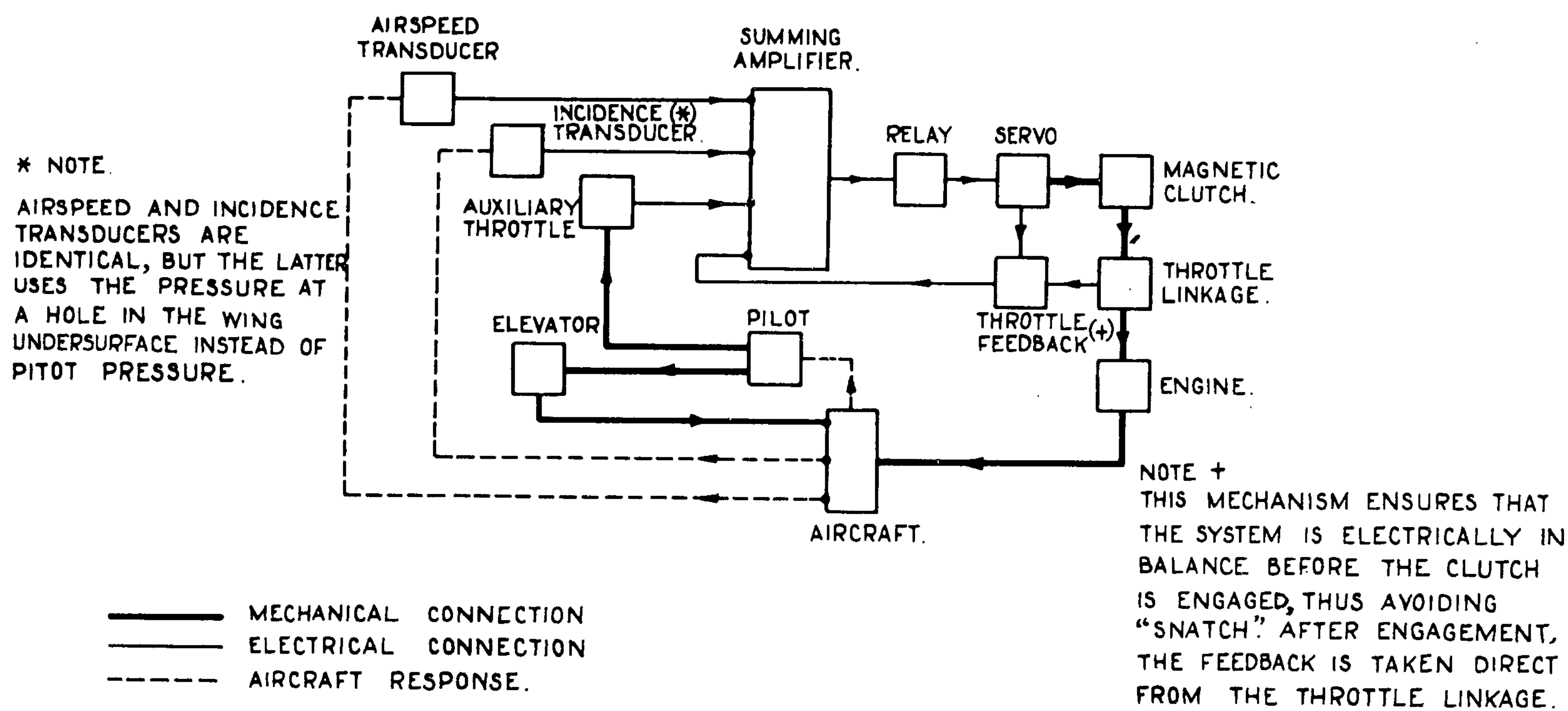


Fig.12 Throttle control system in Avro 707A

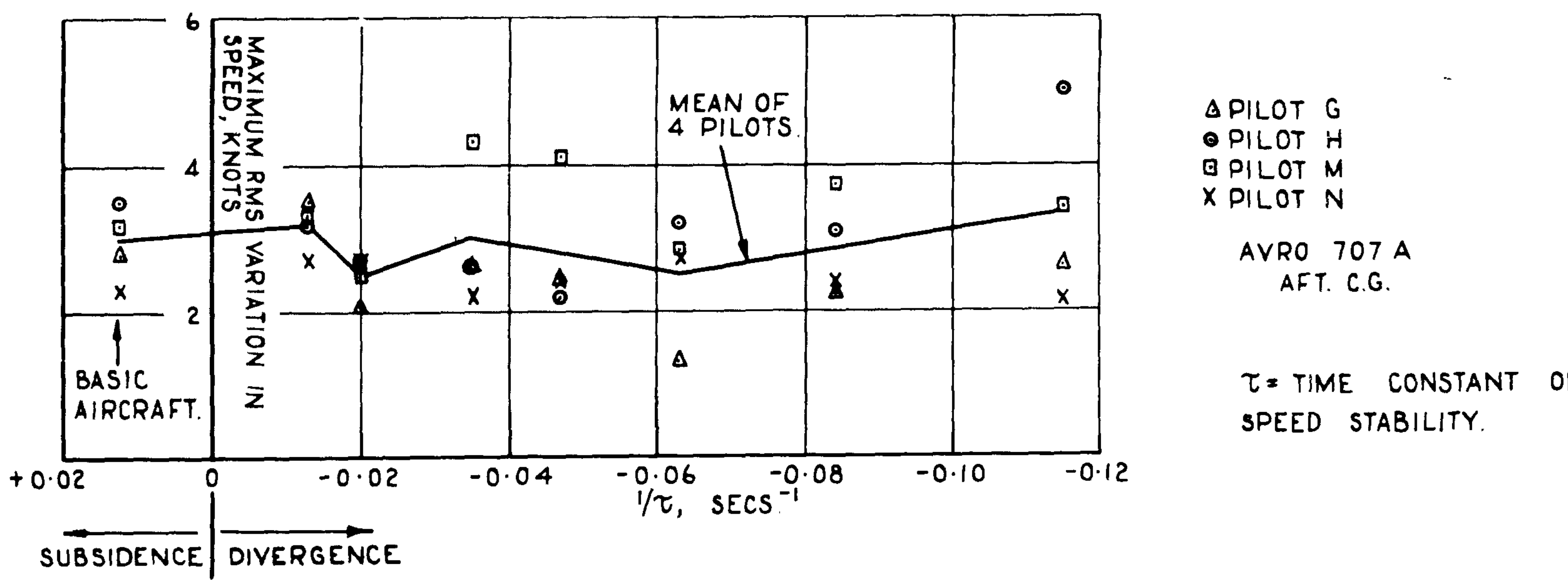


Fig. 13 Effect of speed stability on speed variability (worst results of 6 approaches per pilot)

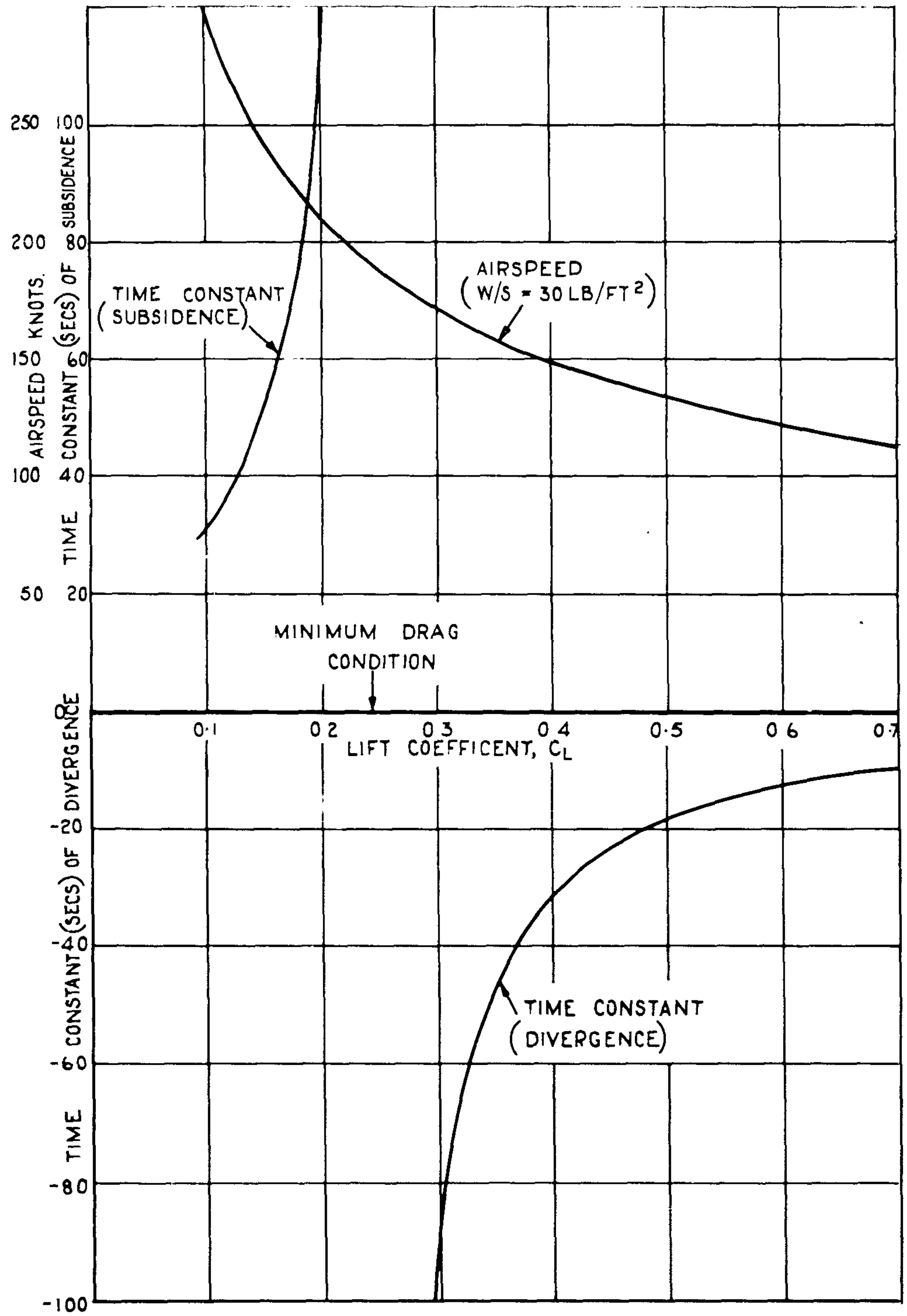


Fig. 14 Speed stability of typical aircraft in approach configuration

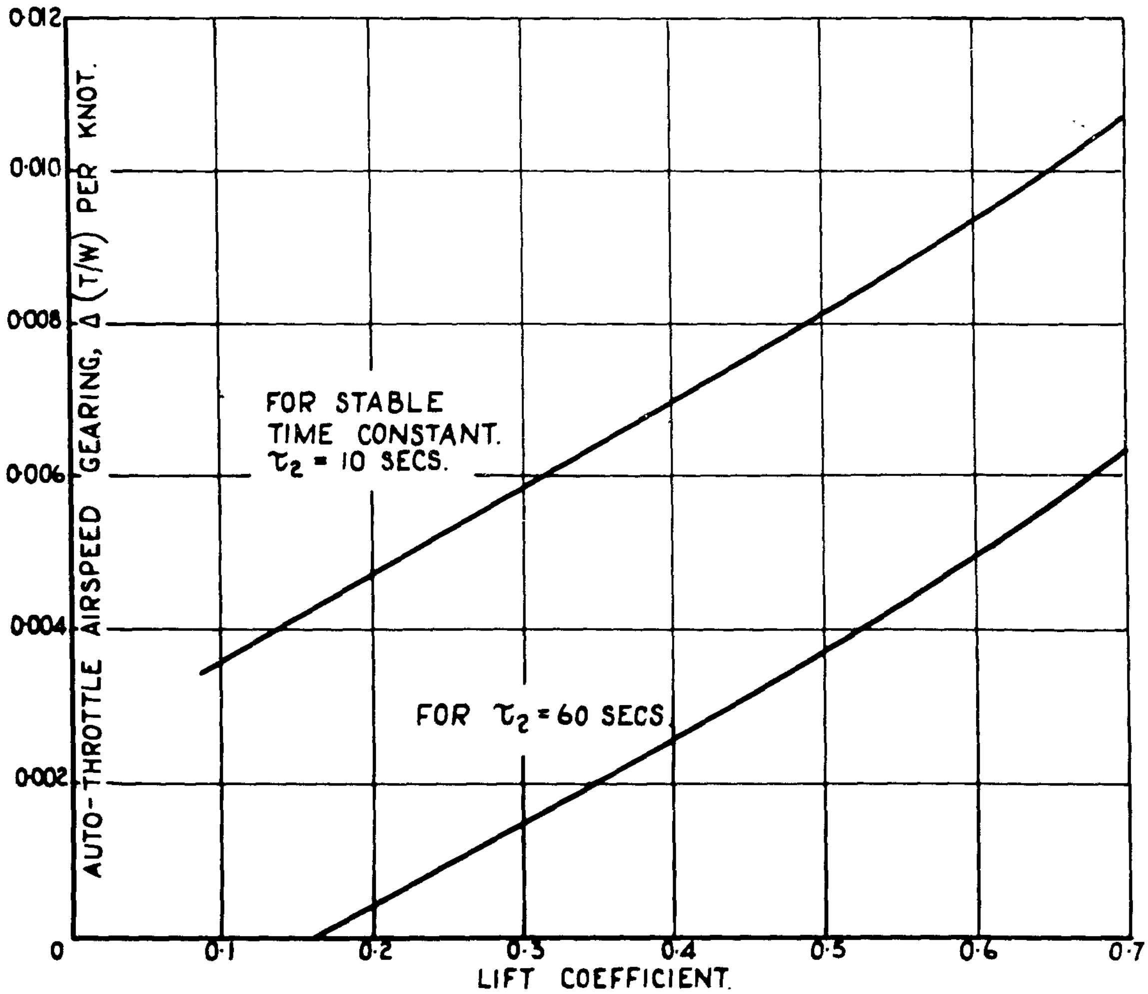


Fig. 15 Auto-throttle airspeed gearings required on typical high-speed aircraft

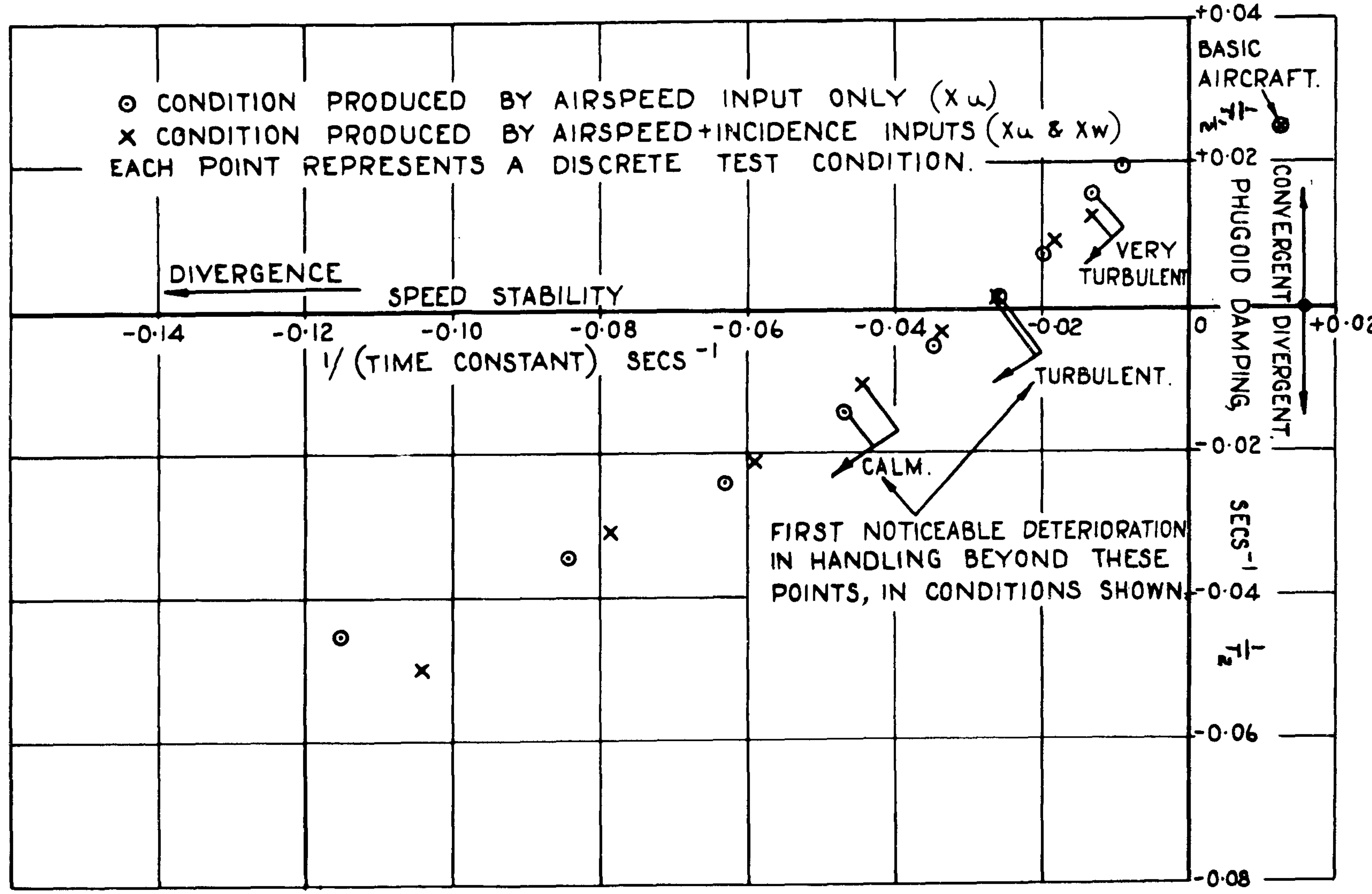


Fig.16 Relation between speed stability and phugoid dampmng in Avro 707A tests

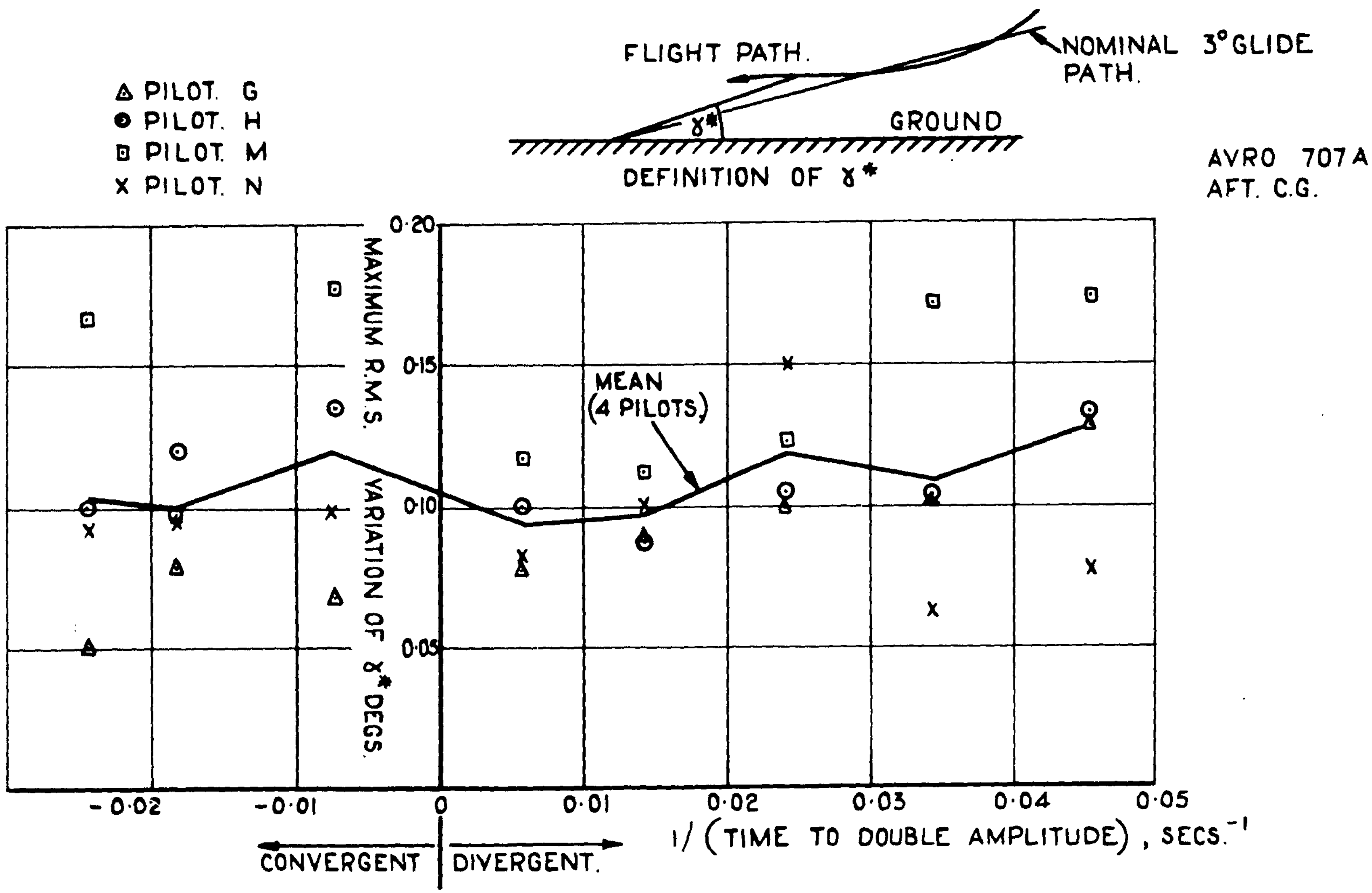


Fig.17 Effect of phugoid damping on glide path holding (worst results of 6 approaches per pilot)

DISCUSSION

J.C. Wimpenny (U.K.): Would Mr. Lean agree that although their average performance was little affected by destabilising auto-throttle there was then more scatter in performance between various pilots?

We are doing some simulator work which is showing that there may be important couplings between the phugoid and the short-period mode. As the latter worsens by either moving the C.G. aft or reducing the S.P.O. damping so there enters in an undesirable excitation of the phugoid even though the phugoid itself is not unstable. This excitation is also sensitive to stick gearing and forces. Has Mr. Lean any views on this, and in particular has he done any work on his research aircraft varying C.G. position which would be expected to show up any such effect?

Author's reply: There is, perhaps, a slight increase in the scatter between pilots, though I doubt whether this is significant - probably less so than the reduction in scatter when the auto-throttle was only slightly destabilising.

Unfortunately, the range over which we can move the C.G. on the test aircraft is too small to produce significant effects on performance of the task. However, Mr. Wimpenny's simulator work is most ineresting. This coupling between the phugoid and short-period mode is, perhaps, not surprising. Poor short-period characteristics result in large excursions from trimmed conditions, and with any normal level of damping of the phugoid, these will take an undesirably long time to decay.

H. Schlichting (Germany): I should like to make some brief remarks on the first part of the paper by A. Spence and D. Lean, which deals with the aerodynamic characteristics of high-lift wings with blowing at the knee of the leading-edge flap and trailing-edge flap. My collaborator, F. Thomas, in the Deutsche Forschungsanstalt für Luftahrt Braunschweig (DFL) has carried out extensive experimental investigations on wings with blowing over the trailing-edge flap (Fig. a). In connection with this he was able to advance a theory of this flow to some extent. It is well known that for wing sections with blowing over the trailing-edge flap the curve of the lift coefficient C_L against the momentum coefficient C_μ is of the type indicated in Figure b. With increasing C_μ there is at first a rather steep increase in the region of 'boundary layer control' and a much smaller increase in the region of 'super-circulation' which, after Spence[1], is nearly linear if C_L is plotted against $\sqrt{C_\mu}$. The value of the momentum coefficient at the border of these two regions $C_{\mu p}$, gives that particular amount of blowing which is necessary just to avoid completely separation of flow at the deflected flap. This value of the momentum coefficient for 'complete boundary layer control' is of considerable practical importance with regard to the efficiency of the high-lift blowing system.

Now Thomas has been able to give a theoretical estimation of this momentum coefficient for 'complete boundary layer control'. It depends considerably on the flap deflection η.

The analysis is based on results of boundary layer measurements in the mixing region of the jet downstream of the blowing slit (Fig. c). Applying the well known approximate method of calculating the turbulent boundary layer with impermeable wall and using the potential flow pressure distribution of the aerofoil with flap deflection, it was possible to estimate the amount of momentum of the jet which is necessary to avoid separation of the flow up to the trailing edge. Of special importance for such an estimation is the knowledge of the losses of the jet due to skin friction downstream of the slit. These losses, which depend strongly on the velocity ratio of the velocities of the jet and the external flow, have been obtained from boundary layer measurements.

This work was done as doctoral thesis of the Technical University of Braunschweig, which is available as DFL-Report[2].

A further remarkable paper dealing with this problem has been published by ONERA[3].

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Captions to Figures

Fig.a Wing profile with blowing over the trailing edge flap; momentum coefficient of blowing C_μ, based on total wing area

Fig.b Lift coefficient C_L against momentum coefficient C_μ for a wing profile of Figure a, at constant values of angle of incidence α and flap deflection η.

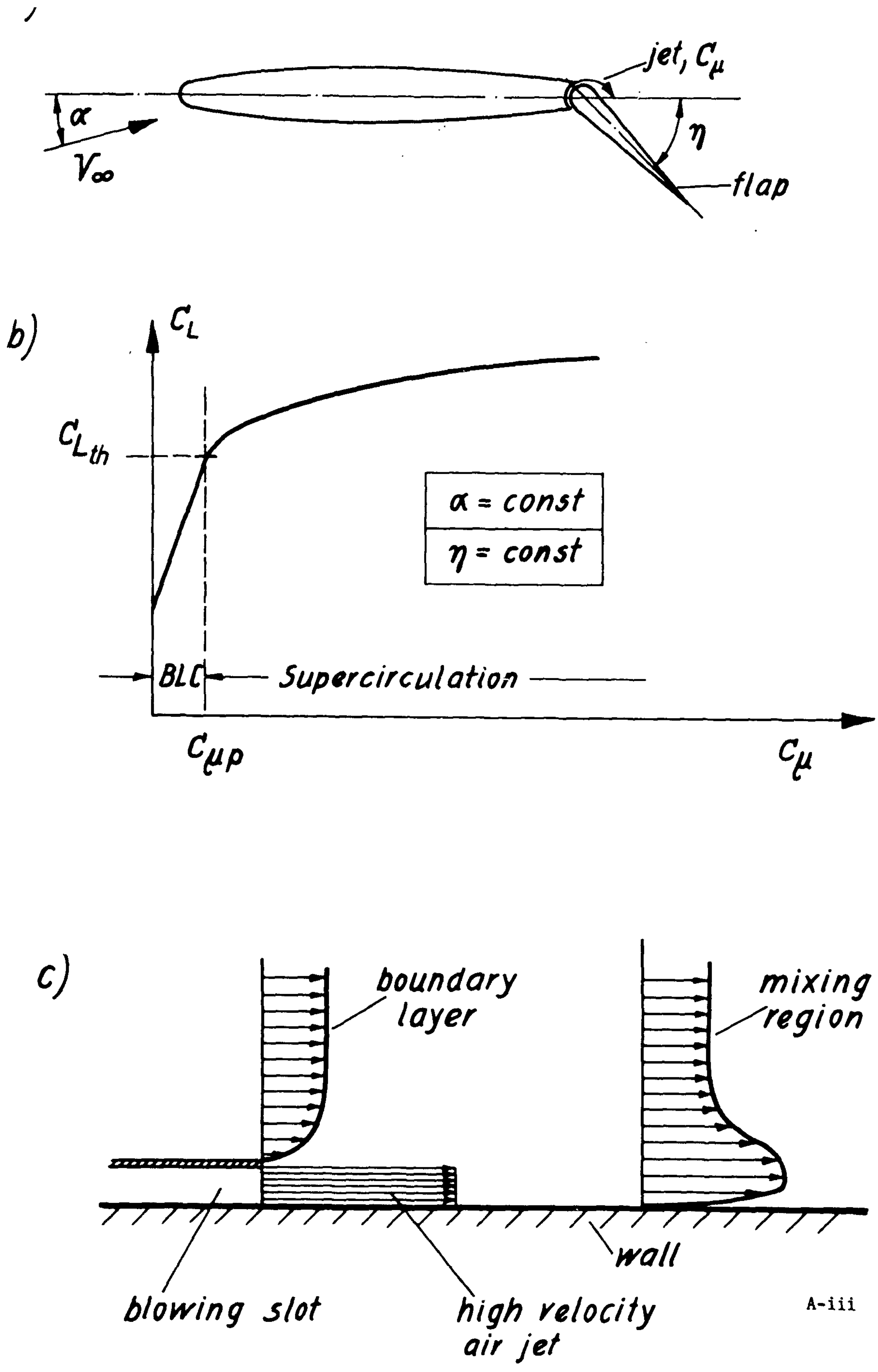
B L C = Boundary Layer Control

$0<C_\mu<C_{\mu_p}$: Boundary Layer Control

$C_\mu>C_{\mu_p}$: Supercircutarion

C_{μ_p} : Momentum coefficient of 'Complete Boundary Layer Control'

Fig.c Turbulent boundary layer with tangential blowing



jet, C_μ
α
V_∞
η
flap
b)
C_L
$C_{L_{th}}$
α = const
η = const
BLC
Supercirculation
$C_{\mu p}$
C_μ
c)
boundary layer
mixing region
wall
blowing slot
high velocity air jet

ADDENDUM

AGARD SPECIALISTS' MEETING

on

STABILITY AND CONTROL

Complete List of Papers Presented

Following is a list of the titles and authors of the 41 papers presented at the Stability and Control Meeting held in Brussels in April, 1960, together with the AGARD Report number covering the publication of each paper.

INTRODUCTORY PAPERS

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AERODYNAMIC DERIVATIVES

AEROELASTIC EFFECTS

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AGARD Report 357
North Atlantic Treaty Organization, Advisory Group for Aeronautical Research and Development
SOME LOW-SPEED PROBLEMS OF HIGH-SPEED AIRCRAFT
A. Spence and D. Lean
1961
35 pages incl. 17 refs. & 17 figs; plus discussion and bibliography of papers presented at the Stability and Control Meeting

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The first part of the paper deals with the low-speed aerodynamics of aircraft shapes suitable for achieving a required range at supersonic speeds. No attention is given to 'slewed' wings, nor to possible application of powered lift or variable geometry. Wind tunnel tests are described on a simplified model with boundary layer control methods applied. Mention is also made of the possibility of adverse ground effect on maximum lift.

P.T.O.

AGARD Report 357
North Atlantic Treaty Organization, Advisory Group for Aeronautical Research and Development
SOME LOW-SPEED PROBLEMS OF HIGH-SPEED AIRCRAFT
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